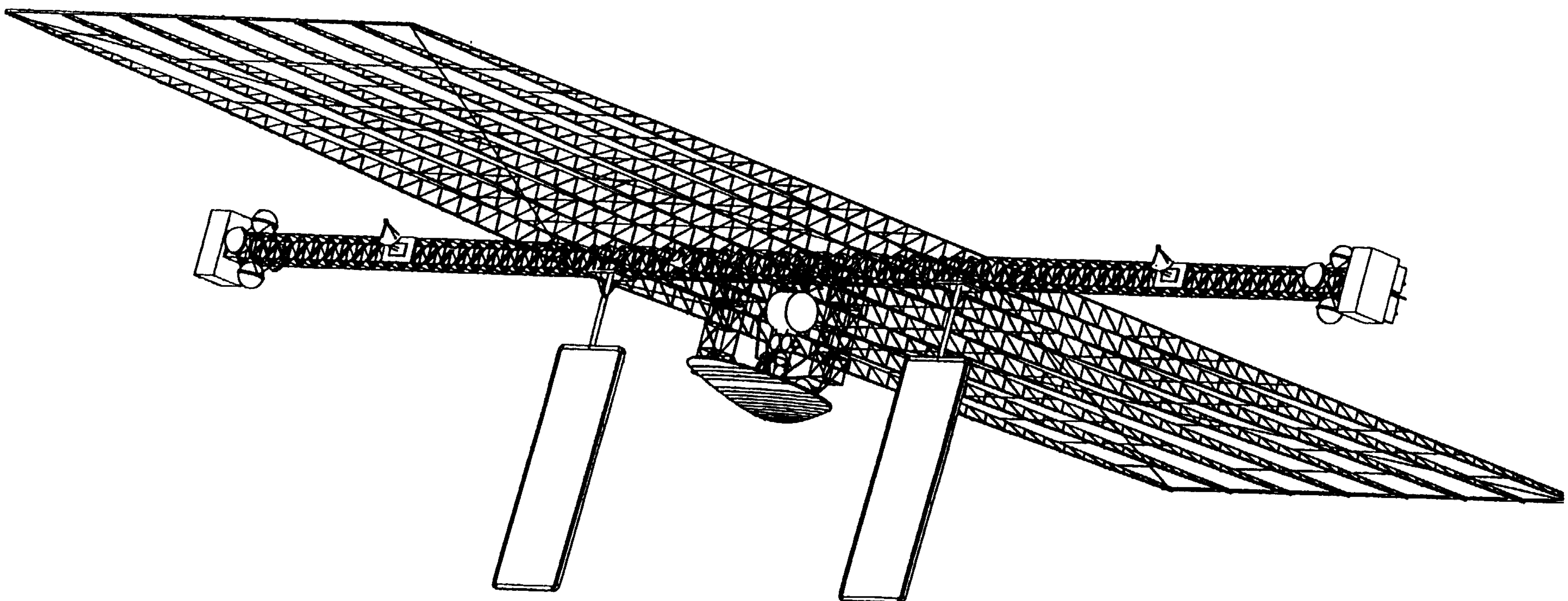
Space Transfer Concepts And Analysis for Exploration Missions



Second Quarterly Review
March 23, 1990

ADVANCED
CIVIL
SPACE
SYSTEMS

Boeing Aerospace and Electronics
Huntsville, Alabama

NASA Contract NAS8-37857

D615-10009

Space Transfer Concepts and Analysis for Exploration Missions

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2nd Quarterly Review

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Boeing Aerospace and Electronics
Huntsville, Alabama

ADVANCED CIVIL SPACE SYSTEMS

D615-10009

CAMUS INCORPORATED
Consultants in Human Potential

GENERAL DYNAMICS
Space Systems Division

MRC

SRS TECHNOLOGIES

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Space Transfer Concepts and Analyses for Exploration Missions

NASA Contract NAS8-37857

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Boeing Aerospace and Electronics
Huntsville, Alabama

Gordon R. Woodcock
Study Manager

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5.0 Radiation Protection (Continued)

10.0 NTR (Continued)

13.0 Orbital and Space-Based Requirements (Continued)

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Acronym List

AB - Aerobrake
ACS - Attitude Control System (same as RCS)
ASE - Airborne Support Equipment
CEP - Circular Error Probability (km)
CG - Center of Gravity
ECCV - Earth Crew Capture Vehicle
ECLSS - Environmental Control and Life Support System
ETO - Earth To Orbit
EVA - Extra Vehicular Activity
FY - Fiscal Year
FSE - Flight Support Equipment
GCR - Galactic Cosmic Radiation
GCR - Gas Core Reactor
GN & C - Guidance, Navigation, and Contol
HLLV - Heavy Lift Launch Vehicle
IMEO - Initial Mass in Earth Orbit
IMLEO - Initial Mass in Low Earth Orbit
IMNSO - Initial Mass in Nuclear Safe Orbit
IR - Infrared
LAD - Liquid Acquisition Device
L/D - Lift to Drag ratio
LEO - Low Earth Orbit
LEV - Lunar Excursion Vehicle
LH_2 - Liquid Hydrogen
LOX/LO_2 - Liquid Oxygen
LRV - Lunar Roving Vehicle
LSSM - Lunar Surface Support Module
LTV - Lunar Transfer Vehicle
MAV - Mars Ascent Vehicle
MEV - Mars Excursion Vehicle
MLI - Multi-Layered Insulation
MMH - Mono-Methyl Hydrozene
MMV - Mars Mission Vehicle
MOLAB - Mobile Laboratory
MSE - Mission Support Equipment
MTV - Mars Transfer Vehicle
NCP - No Current Program
Nd:YAG - Neodymium: Yttrium Aluminum Garnate
NEP - Nuclear Electric Propulsion
NTR - Nuclear Thermal Rocket
OSE - Orbital Support Equipment
RCS - Reaction Control System
SAA - South Atlantic Anomaly
SEP - Solar Electric Propulsion
SPE - Solar Proton Event
SSF - Space Station Freedom
STCAEM - Space Transfer Concepts & Analysis for Exploration Missions
TEIS - Trans-Earth Injection Stage
TMIS - Trans-Mars Injection Stage
TPS - Thermal Protection System
TT & C - Tracking, Telemetry, & Communications
V_e - Entry Velocity (m/s)
WBS - Work Breakdown Structure

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Agenda

Task #	1 Trans. Element/Integ System	2 Trans. Concepts Defination	3 Trades and Sensitivities	4 Integration Compatibility	5 Support Reqts. & Concepts	6 Evolution Assessment	7 Technology	8 Programmatics	9 Special Studies	
Executive Summary										Gordon Woodcock
Reference System (option 5)										
Requirements Data Base	●									
Mission Analysis	●			●						
GN&C, Aeroheating, Landing	●			●						
Radiation	●									Matthew Appleby
Aerobrake Mass Analysis		●								Brent Sherwood
Shuttle-Z TMIS		●	●							
Long Duration Hab Trade Study		●	●	●		●	●			
Alternate Options										
Lunar Evolution, Mini MEV		●	●			●				
NTR update	●		●							Ben Donahue
NEP update	●		●			●	●			Brad Cothran
SEP Update			●			●	●			
Supporting Studies & Tech Directives										Ernie Henshaw
Ops Analysis					●					
Technology Assessment							●			Jerry McGhee
Piloted Rover									●	Gordon Woodcock
Programmatics								●		
Evolutionary/Innovative Architectures						●				

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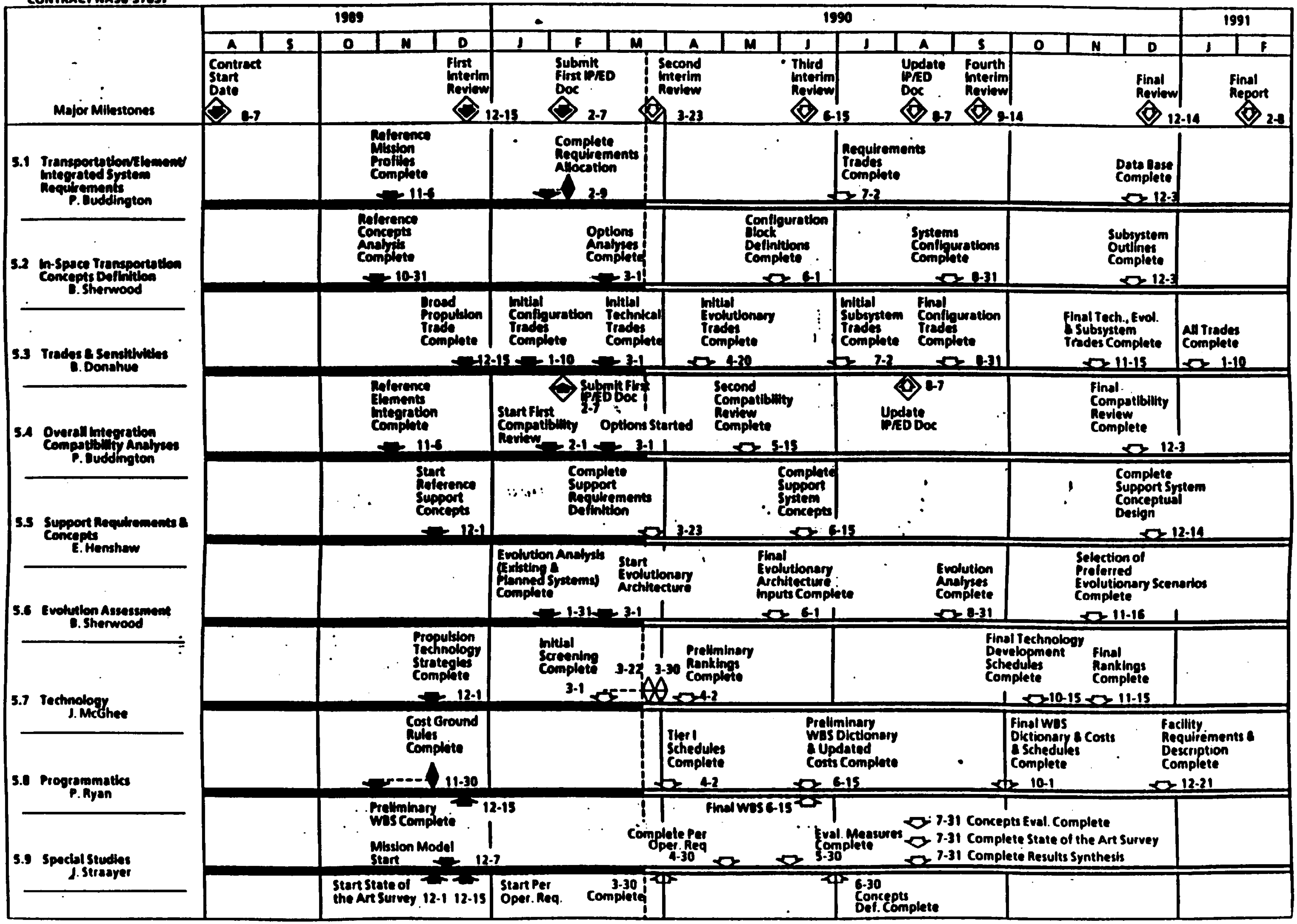
CONTRACT NAS8-37857

SPACE TRANSFER CONCEPTS AND ANALYSES FOR EXPLORATION MISSIONS – TIER I/II SCHEDULE



Program Manager: G. Woodcock
Update Responsibility: C. Hames

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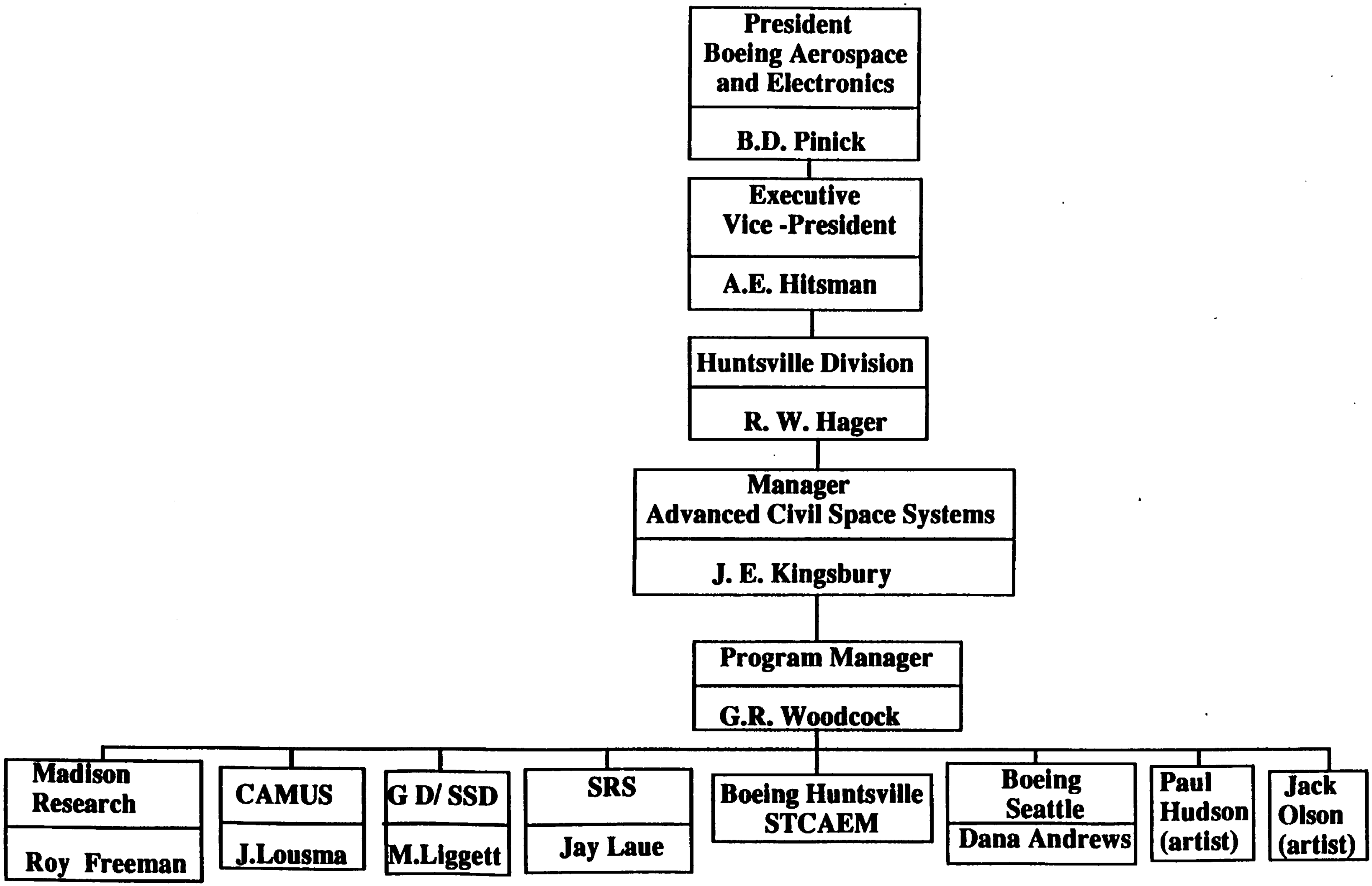
Space Transfer Concepts and Analysis
Boeing Organization



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Study Organization

STCEAM Study Manager
G.R. Woodcock
(205) 461-3954

Task Managers

Task 1.0
Transportation Elements/ Integration Systems
Patricia Buddington
(205) 461-3959
Patrick Ryan
(205) 461-3941

Task 2.0
Concepts
Brent Sherwood
(205) 461-3968

Task 1.6
Support to OEA
Carl Case
(205) 461-3954

Task 3.0
Trades
Benjamin Donahue
(205) 461-3968

Task 4.0
Integration Compatability
Patricia Buddington
(205) 461-3959

Task 9.1
Rover
Joe Straayer
(205) 461- 3953

Task 5.0
Support Requirements
Ernest Henshaw
(205) 461-3956

Task 6.0
Evolution
Brent Sherwood
(205) 461-3968

Task 7.0
Technology
Jerry McGhee
(205) 461-3968

Task 8.0
Programmatics
Patrick Ryan
(205) 461-3941

Other Advanced Civil Space

Keith Stanley, Brian Wallace, John Horton, Joe Straayer, Ed Fisher, Steve Woletz

Matthew Appleby, James Burruss, Stephen Capps, Brad Cothran, Micheal Cupples, Micheal Fouche, Rob Fowler, Patoric Hoy, Stephen LeDoux, Jill Nordwall, E.R. (Bud) Tanner, Brian Tillotson

Space Station Support

Dani Eder, Susan Doll, Brand Griffin

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Executive Summary

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SYSTEMS

Agenda

2nd Quarter Findings
Requirements Summary
Aerobrake and Landing Summary
Hab Trade Summary
Advanced Propulsion Summary
Evolutionary Architecture Summary

Study Objectives Review and Status

The facing page summarizes our current status in meeting the objectives of the contractual Statement of Work. (Ellipses in the statements of objectives indicate omissions for brevity.) The Statement of Work objectives were mainly satisfied for the cryogenic/aerobraking reference system during the NASA "90-day Study". Current work is directed to satisfying them for the other space transfer options.

Study Objectives Review and Status

Statement of Work Objectives	Status
1. Assess and critique the NASA-provided mission model(s)/mission descriptions/scenarios; develop in-space system requirements for each scenario.	Completed Options 1 and 5. Now starting to devise and analyze new options.
2. Define and assess in-space transportation concepts to the subsystem level ... including support systems.	Completed Options 1 and 5. Now starting to define and assess new options.
3. Define and assess habitability conceptual elements ...	Complete except for updates as required.
4. Examine, characterize, and compare different alternatives; assess commonality; recommend/ define preferred concepts and configurations.	Completed first iteration on advanced propulsion systems; now starting second iteration on alternative architectures
5. Assess all operations associated with candidate concepts....	Doing operations analysis for Option 5, and beginning analysis of other options.
6. Develop programmatic data ...	Completed 3rd iteration on Option 5 cost.
7. Coduct comprehensive trade studies ... and select best overall concepts.	Many trades complete; many more in progress.
8. Perform overall integration compatibility analyses ...	Complete for Option 5; continuing for intercompatibility of missions & elements.
9. Identify and prioritize enabling and enhancing technology requirements ...	Completed initial technology requirements assessment against Option 5 and alternatives.

Mars Mission Vehicle in LEO

The Mars vehicle LEO configuration is shown here ready for trans-Mars insertion (TMI).

The TMI stage launches the vehicle out of Earth orbit on a trans-Mars trajectory. There are four propellant tanks and five engines in the TMI stage; it is modularized for compatibility with the launch vehicle. The elements of the TMI stage are launched fully loaded with propellant.

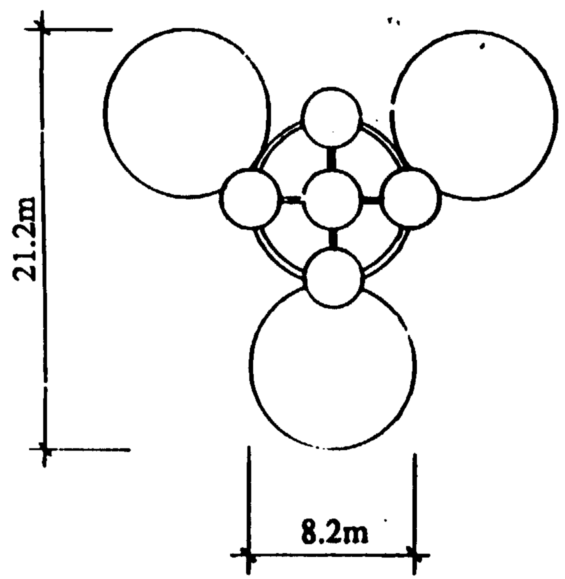
The Mars excursion vehicle includes an aerobrake for Mars capture and entry/landing, a descent propulsion stage, an ascent propulsion stage with crew module for Mars descent, ascent, and contingency surface operations, and 25 t. of surface payload (a habitat and science) for normal surface operations.

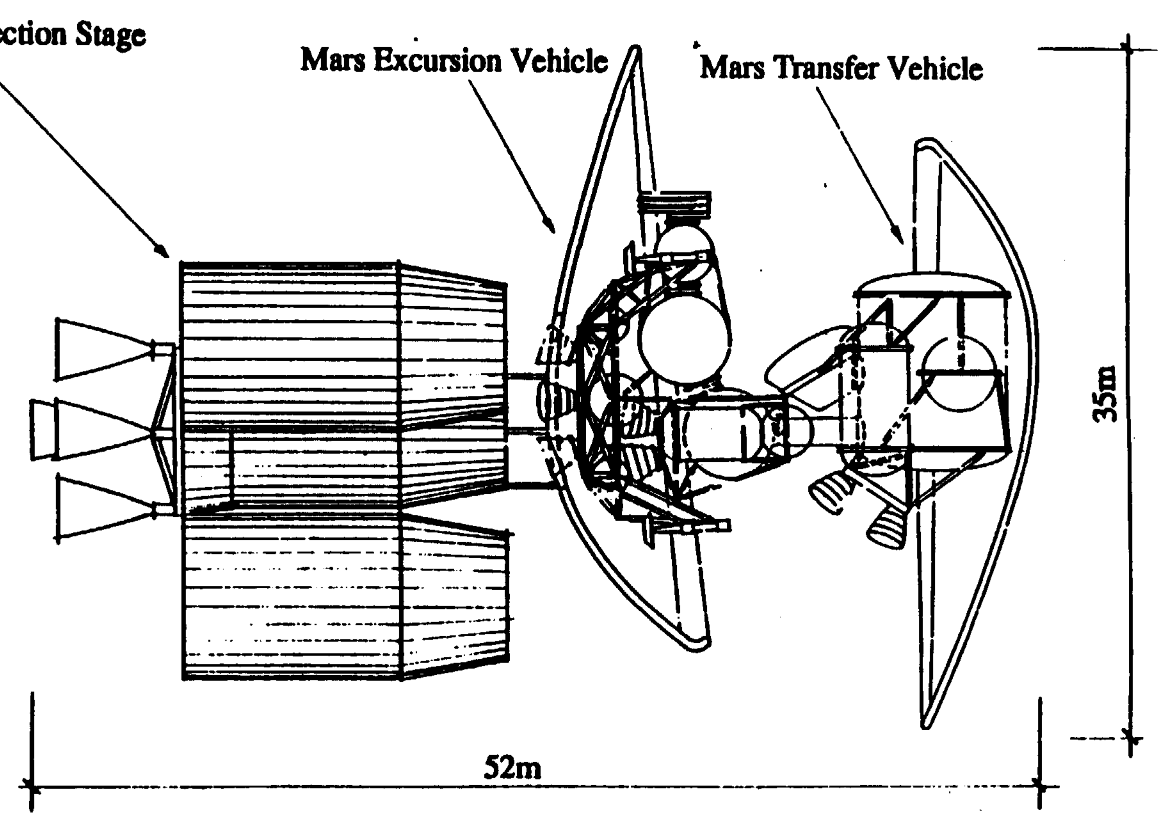
The Mars transfer vehicle includes its own aerobrake for Mars capture, a long-duration crew habitat for the trips to and from Mars, a propulsion system for boost out of Mars orbit to return to Earth, and the Earth crew capture vehicle. The TMI stage is bookkept as part of the Mars transfer vehicle for WBS purposes. On some missions, the MTV aerobrake returns to Earth with the vehicle so that the MTV (except for the TMIS) can be captured in Earth orbit for reuse on another mission.

All crew volumes are contiguous between the MEV and MTV during TMI and coast.

The mass totals for option 1 and 5 are shown for comparison. The only difference between options 1 and 5 is that option 5 carries a surface reconnaisance vehicle into Mars orbit on the MEV (it is not shown on the chart). The surface reconnaisance vehicle is launched from the Mars parking orbit to perform robotic exploration of a future human landing site.

Mars Mission Vehicle in LEO





Mass for Option 1	
MTV	138.7 t
MEV	79.0 t
TMIS	509.8 t
Total IMEO	727.5 t

Mass for Option 5	
MTV	138.7
MEV	83.6 t
TMIS	517.1 t
Total IMEO	739.4 t

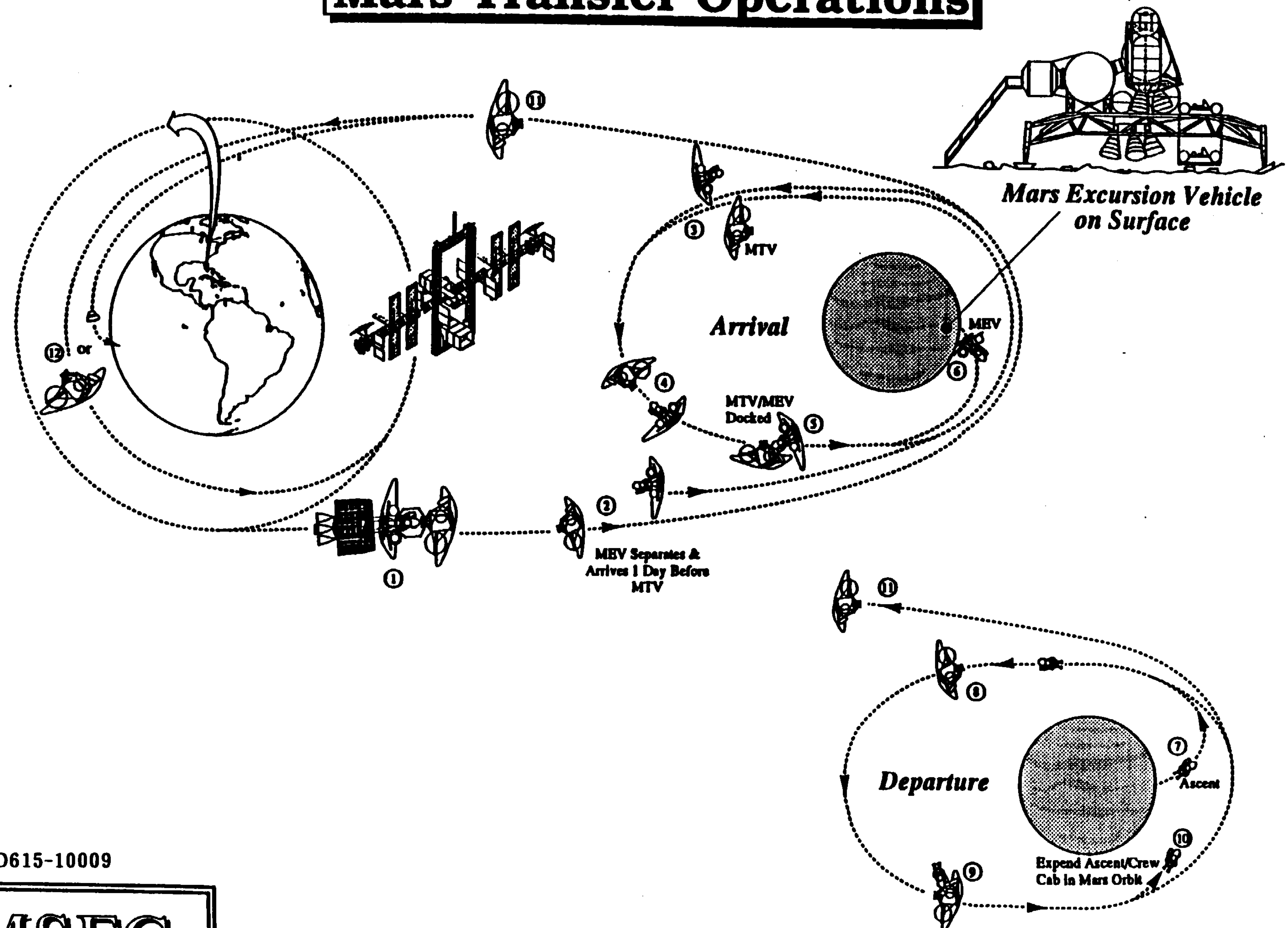
Mars Transfer Operations

Mars transfer operations for the reference system are illustrated here. The opposition profiles normally include a Venus swingby either going to Mars or returning to Earth. Occasionally, a Venus swingby may be used each way. (The reference 2015 mission uses an outbound swingby; there is also an alternate inbound swingby profile for this opportunity.) Venus swingbys are normally unpowered and there are no operational events at the swingby. The nominal mission sequence is as follows:

1. Reference Cryo/AB Mars vehicle leaving Earth orbit.
2. MEV/MTV separate 50 days from Mars.
3. Unmanned MEV captures into Mars orbit 1 day prior to MTV.
4. MTV/MEV rendezvous and berth in Mars orbit.
5. Crew transfers from MTV to MEV.
6. MEV descends to the surface of Mars.
7. MAV ascends from surface, leaving descent stage.
8. MAV/MTV berth in Mars orbit.
9. Crew transfers from MAV to MTV.
10. MAV left in Mars orbit.
11. MTV departs from Mars toward Earth.
12. MTV captures in LEO, or crew returns to Earth's surface in ECCV.

Acronymns: AB - aerobraking; MEV - Mars Excursion Vehicle; MTV - Mars transfer vehicle, includes trans-Mars injection stage as well as transfer propulsion and hab; MAV - Mars ascent vehicle; LEO - low Earth orbit; ECCV - Earth crew capture vehicle (like an Apollo command module).

Mars Transfer Operations



Principal Findings Dec. 1989

This page summarizes results presented at the first study review. Definition of a baseline cryogenic/aerobraking system that could perform all of the mission opportunities of interest required a joint and cooperative analysis of vehicle parametrics and performance, and of mission profile options. The first three items reported were accomplished by a joint effort by Boeing and its subcontractors, MSFC, and the MASE mission analysis team.

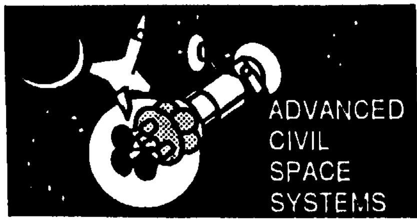
By iteration of the design, we satisfied not only the identified design requirements, but desirable features. Additional findings for the reference system include a family of aerobrake shapes that permit tailoring over the L/D range 0.5 to slightly more than 1.0, and that cryogen boiloff can be reduced to acceptable levels with passive thermal insulation design. Our present estimates are that the cryogenic system offers minimum mass up to at least 600 days' surface stay. Modest improvements in storable propellant performance, such as through metallic additives, could reverse this conclusion at the longer stay times.

We analyzed launch manifesting in with MSFC and the General Dynamics STIS study managed by MSFC; conclusions are indicated. We considered a range of payload capabilities from 90 to 140 t. and a range of shroud sizes from 8 to 12.5 m. diameter.

Indications from our commonality and evolution analyses are that there is much pay dirt here yet to be exploited.

We performed extensive low-thrust trajectory and mission profile analyses, concluding that electric propulsion systems can be used for Mars crew transportation, with competitive trip times.

The high-Isp NTR item was described on the previous page. The technology advancement strategy is presented later in the briefing. Our conclusion was that an advanced propulsion technology path can be chosen befor full-scale development funding.



Principal Findings Dec. 1989

- Mission profiles can be defined so that an "umbrella" delta V budget captures all opportunities of interest. (This result obtained in cooperation with MASE mission analysis.)
- Reference cryogenic/aerobraking vehicle system supports all opposition opportunities 2010-2020 and all cargo and conjunction opportunities.
- Free return aborts possible most opportunities; vehicle design supports powered Mars flyby abort when necessary.
- Reference conceptual design meets requirements and "desirable features" such as contiguous pressurized connection of all crew modules.
- Aerobrake shapes selected from family of shapes; permits tailoring of aero characteristics. Reference concept L/D 0.5; indications that Mars excursion vehicle may need L/D >= 1.
- Boiloff can be reduced to acceptable level with passive design, including Mars surface up to 600 days.
- Manifesting reaches 75% - 85% efficiency; enhanced by "shuttle-Z" approach.
- Great commonality potential, yet to be fully exploited in cost analyses.
- Advanced propulsion options offer faster trips, reduced mass, or both. Early cost indications indicate that any advanced propulsion option must displace other developments to be economically attractive. (Reduced mass is not enough.)
- Analysis of low thrust propulsion and trajectory options shows trip times competitive with cryogenic/aerobrake reference.
- Solid-core NTR may be able to achieve Isp > 1200 seconds at low chamber pressure (low thrust). This offers significant performance advantages over the 900-second range.
- Technology advancement strategy for advanced propulsion indicates selection in 6 - 8 years, before high-rate funding, if technology programs are implemented. GCR may take a little longer.

Second Quarter Findings, March 1990

The main findings of the second quarter are presented here. Continuing analyses of high-energy aerobraking are revealing second-order problems that typically arise when designs using challenging technology are penetrated to greater depth. The issue of Mars atmosphere uncertainty is presently being addressed by a NASA working group. We are pursuing the sensitivity of GN&C schemes to different kinds of atmosphere uncertainties. We are investigating the radiation problem, cooperating with Ames Research Center specialists.

The technical advantages of advanced propulsion include reduced initial mass, reduced resupply mass, greater reusability, faster trip times, and greater flexibility in mission design. The high L/D requirement for landing, for example, is alleviated for advanced propulsion systems because they have greater flexibility in orbit selection. Whether the technical advantages of advanced propulsion translate into cost advantages will be addressed in the next few months of study effort.

Completion of our habitation module trade resulted in a crew module evolution and commonality scheme applicable to the entire Space Exploration Initiative program.

We have begun to synthesize evolutionary architectures. The ones presented later in this briefing aim at a program strategy of early goal accomplishment and evolution to robust, highly productive systems.

Second Quarter Findings, March 1990

- Complications of aerobraking are surfacing:
  - Mars atmosphere uncertainty and GN&C.
  - Higher L/D required for landing - because cryo/aerobraking parking orbits at Mars should be optimized for minimum delta V, landing at a permanent base site requires cross-range.
  - Severe radiation heating peaks at Mars for massive MTVs.
- Advantages of advanced propulsion becoming more attractive:
  - Refining mission profiles for all.
  - Gravity assists for low-thrust
    - SEP trip times about 500 days.
    - NEP trip times less than 400 days.
  - Resupply/turnaround LEO mass significantly less then IMLEO.
  - Reusable modes.
- Major hab trade shows family of Space Station Freedom derivatives for first decade; add large diameter (~ 8 m) design for second decade.
- Evolutionary architecture analysis shows several promising architecture alternatives
  - Potential for early Mars trip.
  - Novel mission modes.
  - Extensive potential for commonality at major subsystem level.
  - Significant advantages for advanced technology; ambitious lunar/planetary activity levels can be supported.
  - Opportunities for synergistic evolution of architectures and technologies.

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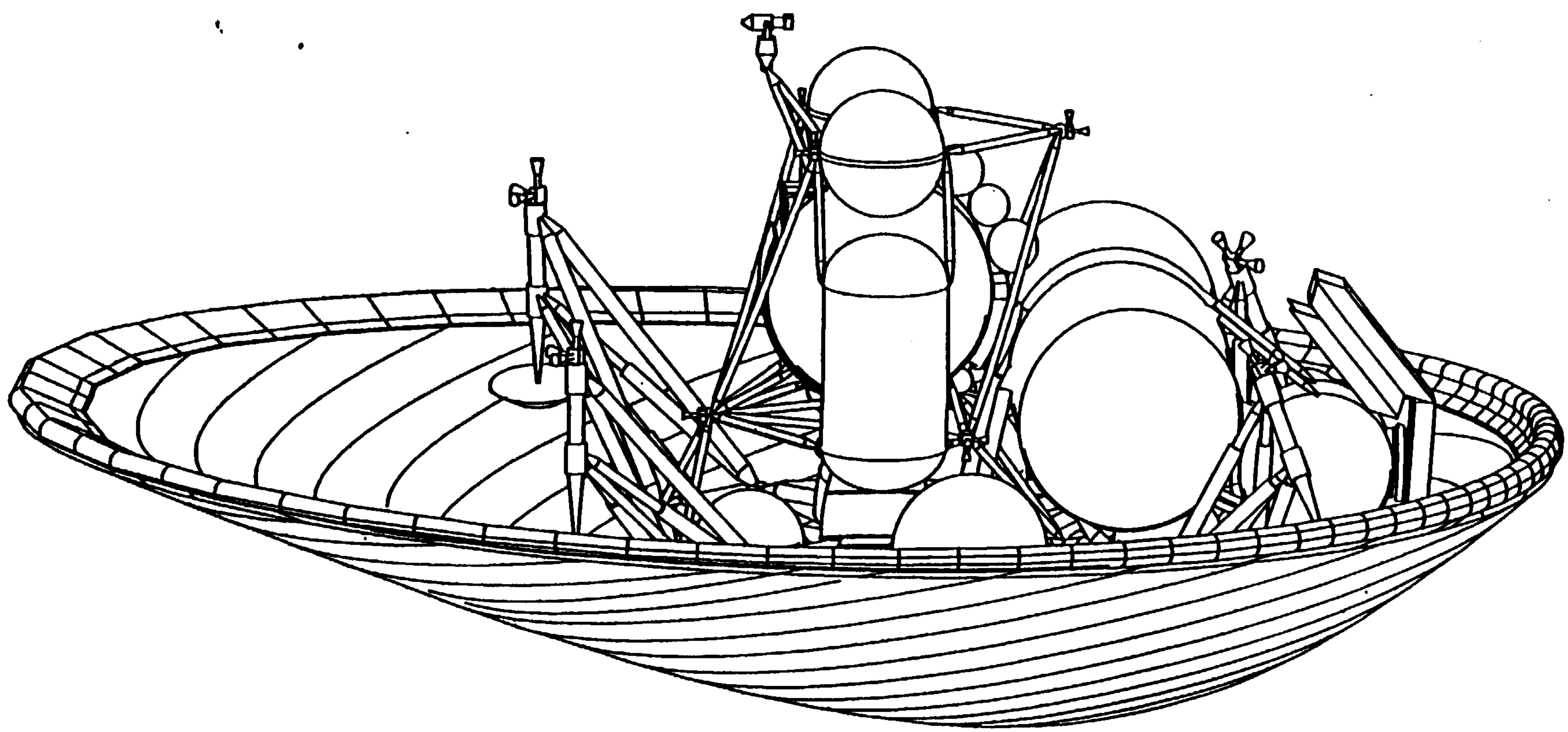
MEV/Aerobrake Configuration



MEV Aerobraking Constraints

The aerobraking constraints applied to the MEV configuration are summarized on the facing page. These constraints include center of mass location for trim at the desired L/D, keeping the MEV itself within the protected wake region, and positioning the crew module for best visibility during aerobraking and powered descent.

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MEV Aerobraking Constraints

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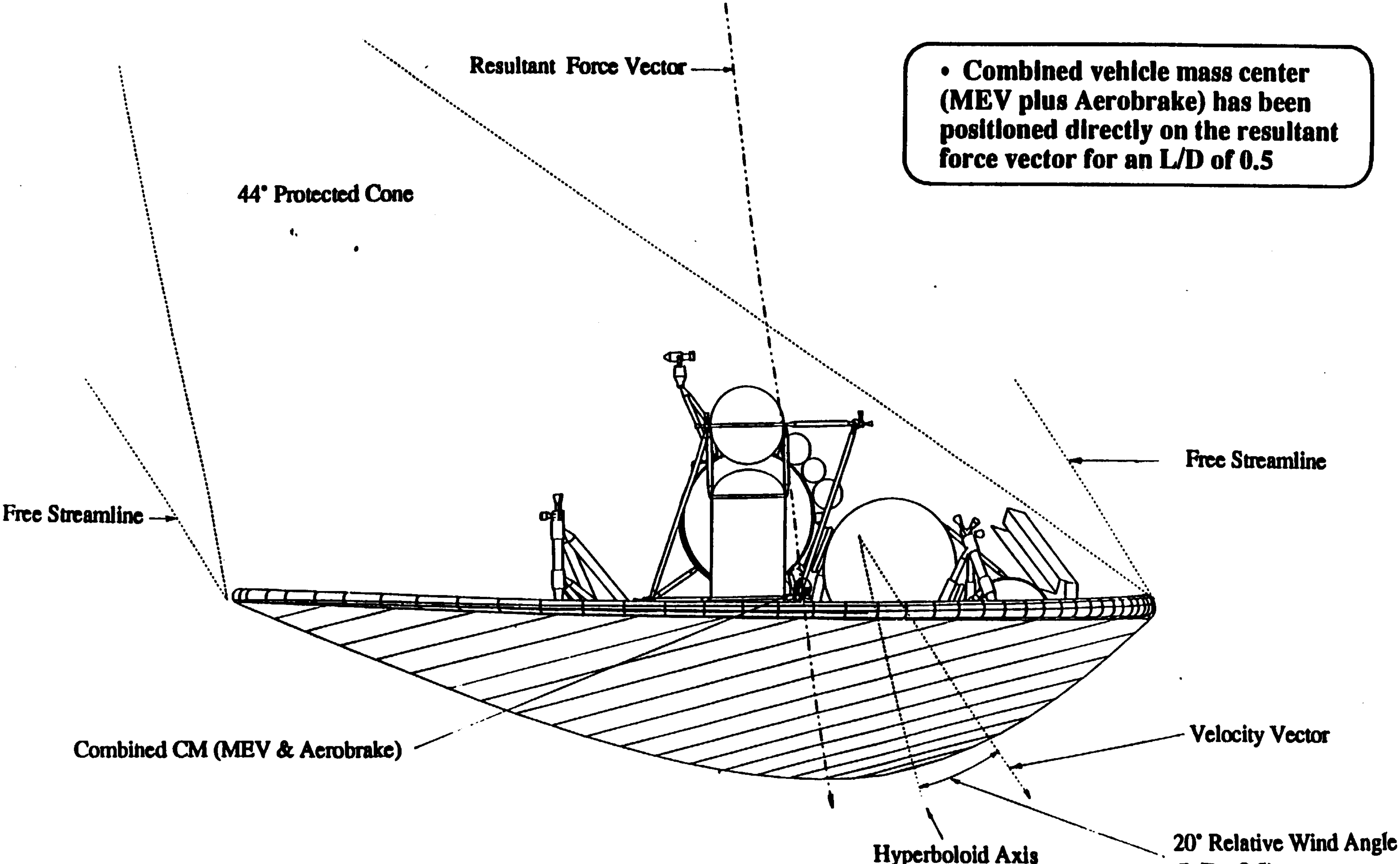
- **Combined vehicle mass center (MEV plus Aerobrake) has been positioned directly on the resultant force vector for an L/D of 0.5**



Aerobraking and Landing Summary

The facing page summarizes findings for guidance, navigation and control (GN&C), aeroheating, and landing L/D performance.

Our reference aerobrake provides a trim L/D of 0.5 and uses roll modulation for flight control. Altering the L/D for this class of shapes requires very large highly-loaded flaps or excessive consumption of attitude control propellant. The nominal trajectory design is illustrated here. The vehicle points the lift vector to the left, then to the right, and then back to the left, flying a dog-leg path that ends up very near the initial heading. The amount of lift exerted in the vertical plane is about ± 0.2; this suffices to compemsate for expected variations in atmosphere density. The slight irregularities in the vertical path result from finite roll rates; as the vector is rolled from one side to the other, near maximum L/D is briefly exerted in the vertical plane.

Predictions of aeroheating using correlations provided by Dr. Chul Park of Ames Research Center (ARC) show severe radiation peaks near maximum dynamic pressure. (We have recently become aware of other correlations, pointed out by Mike Tauber of ARC, that predict less heating.) There are several mitigating strategies as noted on the chart.

Landing footprint studies indicate a need for a Mars descent L/D of 1 to 1.2, if the Mars capture orbit is optimized for interplanetary delta V requirements. This need is premised on a requirement to land anywhere in a ± 20° Mars latitude band on any cryogenic/aerobraking opposition mission opportunity. Advanced propulsion, which has greater flexibility for tailoring the parking orbit, use of conjunction opportunities, or use of low circular Mars parking orbits may mitigate the requirement. On the other hand, if the latitude band requirement is enlarged, still higher L/D might be required.

Aerobraking and Landing Summary

- Roll-modulated trajectory designs for L/D 0.5 and range of C3s and atmosphere densities.

- Exploring effect of atmosphere gravity wave dimensions on guided trajectory exit errors; Δv penalties.

- Exploring two GN&C schemes: (1) redesign trajectory in real time; (2) adaptive guidance with gain schedules adjusted based on atmosphere effects.

- Predicted aeroheating at Mars shows severe radiation heating.

  - Reduce C3 limit to ~ 30; delta V budget impact some years.
  - Fly "lift down" centerline, but risks skip-out
  - Change shape to reduce shock standoff distance; higher L/D and probably greater mass.
  - 2600-3200° K class materials; cost, mass impact?

- Need L/D about 1 for landing unless advanced propulsion can tailor parking orbit.

- Aero flare on approach reduces landing delta V and thrust requirement, but requires pitch control; roll modulation won't do it.

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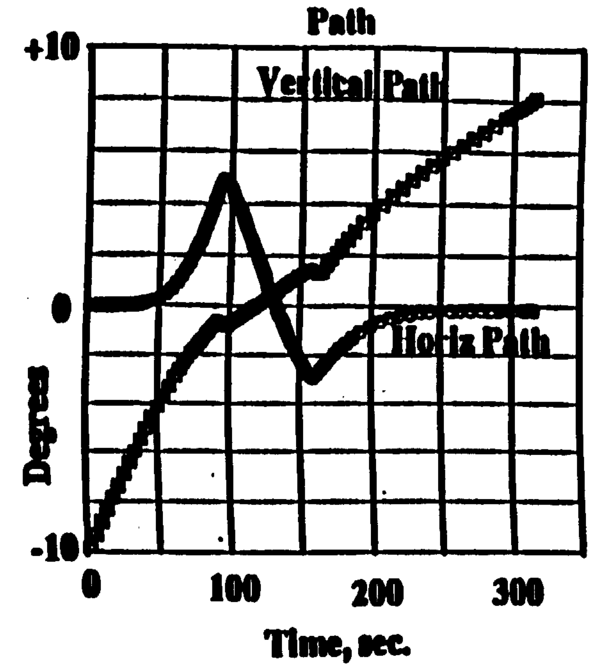
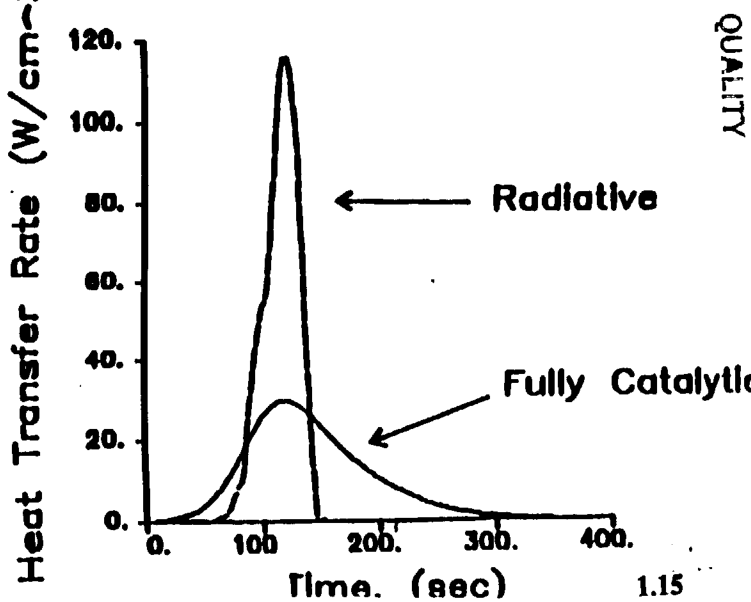




Long-Duration Habitat Trade Study Summary

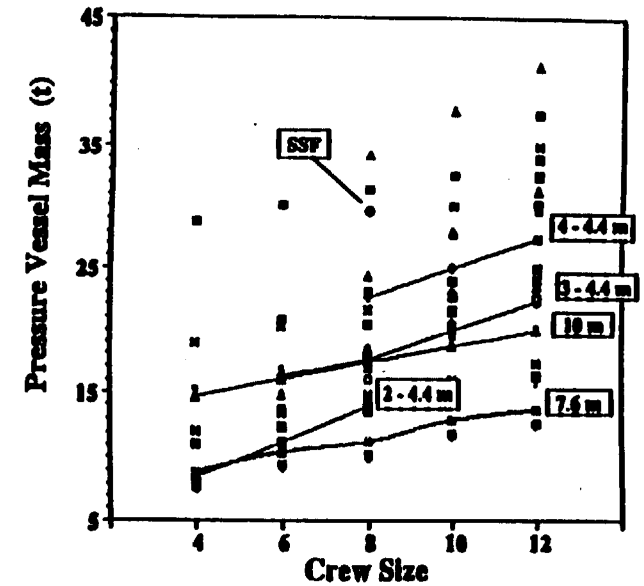
The habitat trade is summarized here. This trade concentrated on the Mars Transfer Vehicle habitat, because it must support a crew for almost three years in the worst-case conjunction mission abort case, and supports the crew for more than a year in most mission profile cases. This habitat can also be used as an advanced lunar surface or Mars surface habitat.

The selected concept uses a 7.6-meter diameter, which traded as clearly superior to the other diameters considered from the mass standpoint, and was evaluated as competitive according to the other evaluation criteria.

Long-duration Habitat Trade Study Summary

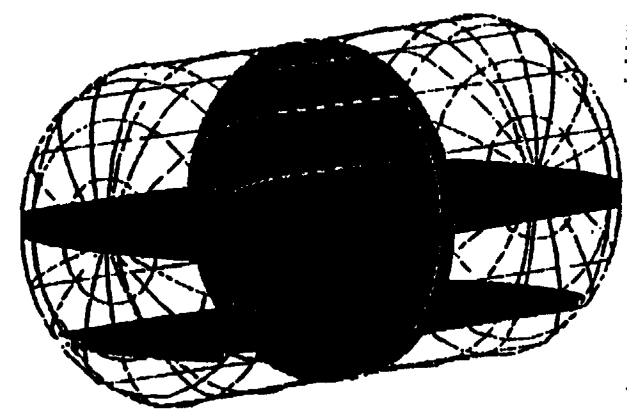
Process

- Trade space matrixed 5 crew sizes and 3 module sizes
- Generated 1480 distinct options, based on gravity, orientation, topology and structure; focused on 150 concepts
- Developed metrics for selecting preferred topologies and geometries; reference configurations for crew response survey
- Weighed pressure vessel structures, estimated equipment outfitted weights; assessed integration impact, commonality, growth potential, manufacturing options



Results

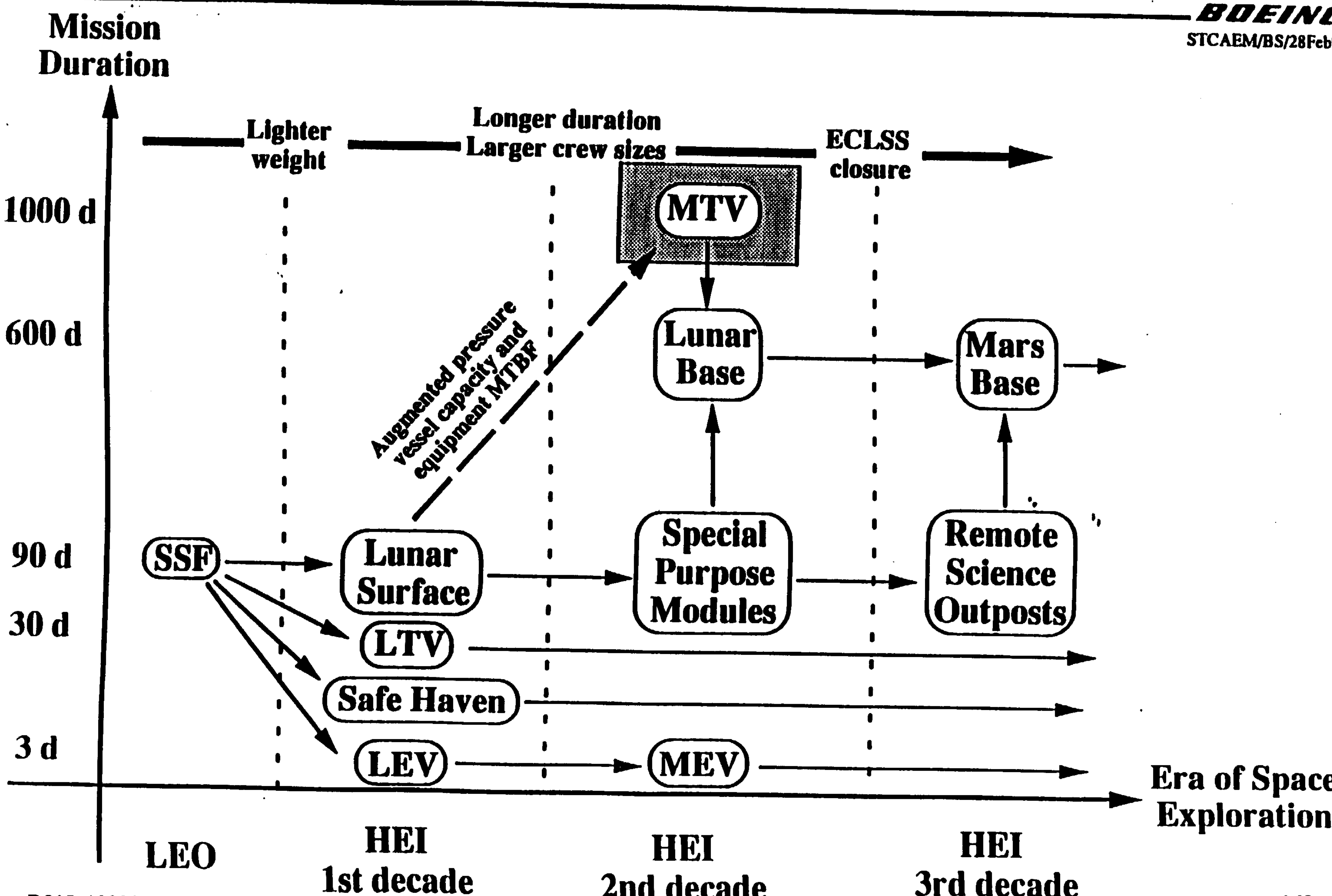
- Generated data allow applying a wide variety of priority sets to determine "optimal" concepts for specific architectures
- First HEI decade can use lightened SSF derivatives for all crew systems: LTV, LEV, surface outposts, safe-havens
- Later, long-duration missions require a larger module, common across architectures and capable of integration with smaller modules
- Trade neckdown led to synthesizing novel module concept, using best features from the studied options
- A 7.6 m diameter vessel, "tunnel-oriented", sized for 6 crew, with a cross-sectional bulkhead, was selected as the reference modular unit



Habitat Module Evolutionary Context

A complete evolution strategy for crew modules is depicted here. The MTV hab which was the subject of the detailed trade study is shown at the top of the chart in a shaded box. The direct Space Station Freedom derivatives are all SSF diameter, with lighter-weight end domes, and with subsystem complements tailored to mission requirements. Short-duration modules, for example, optimize with mainly open-loop life support.

Habitat Module Evolutionary Context



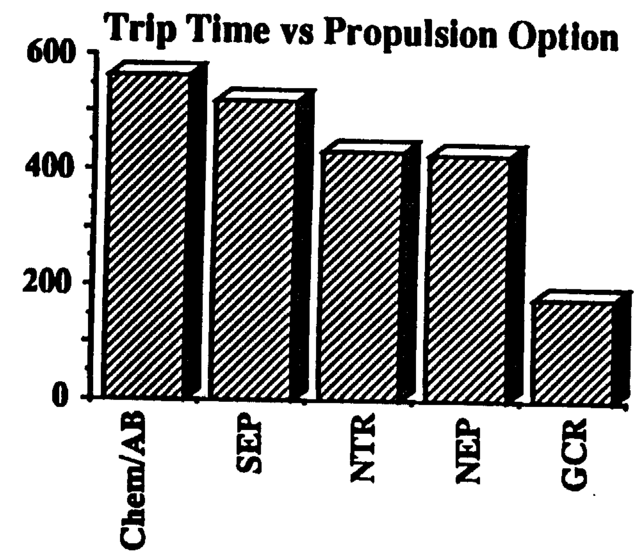
Advanced Propulsion Summary

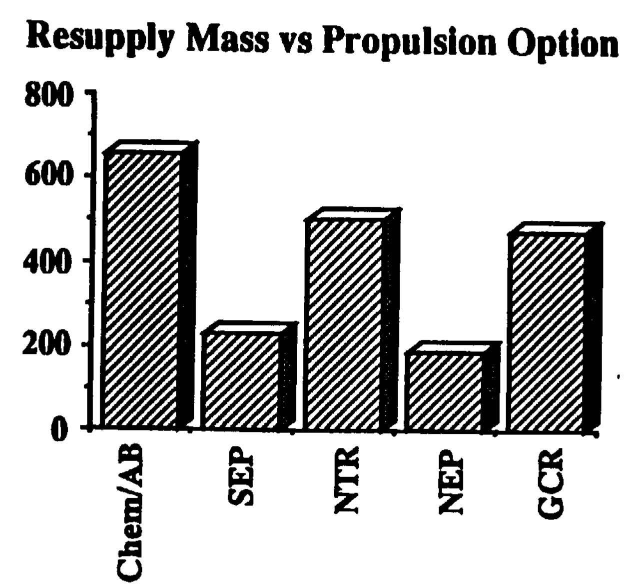
Findings since the December 1989 review are summarized on the facing page. Trip time results have not changed much, except that we have reduced low-thrust trip times somewhat by use of gravity assists. On this chart we show resupply masses for the propulsion systems rather than initial mass in Earth orbit (IMLEO). The low-thrust systems have a significant advantage in resupply mass because most of their mass is reusable propulsion system and payload. A life cycle average, considering initial placement and occasional replacement of limited-life hardware would add some to the mass comparison, but the low-thrust systems will still show significant advantage.

The gas-core rocket, if operated at mission durations typical of the low-thrust systems, would be very competitive in resupply mass.

Advanced Propulsion Summary

- Advanced propulsion options being worked in architecture context; missions and LEVs, MEVs, etc, tailored to their strengths.
- Gravity assists reduce trip time for SEP and NEP; ~500 day trips for SEP, <400 days for 40 MWe NEP.
- SEP can be efficiently transported to L2 node by transfer array.
- Important to use resupply mass rather than IMLEO for electric propulsion evaluation. (Resupply < IMLEO)
- Range of NTR configuration concepts developed.
- NTR all-propulsive performance advantage becomes compelling at Isp ~ 1100 - 1200; at 900 it is mainly a backup to cryogenic/aerobraking.
- Thrust-to-weight less important than Isp; T/W as low as 0.03 works well with multi-burn departures. Low T/W drives burn time per mission to several hours.
- Nuclear-safe orbit not an attractive option (debris; differential nodal regression). We recommend SSF orbit after a cool-down period of several months.
- Advanced propulsion operations need focussed analysis.





Evolutionary/Innovative Architectures Summary

We have identified seven alternative architectural options, and performed preliminary analysis of one of them. The conclusions here are tentative, based on that preliminary analysis. We found a potential to support up to dozens of people on the Moon and Mars at a launch rate of 6 to 7 100-tonne HLLVs per year, by applying advanced space propulsion technology, in-situ resources to minimize resupply requirements, and in-situ resources for transportation propellant.

Evolutionary/Innovative Architectures Summary

- Identified 7 alternative architectures, copvering the advanced propulsion options and major mission profile options.
- Advanced technology provides high leverage for expanding program activity level.
- Great potential for broad commonality of major system modules.
- Indications point to cost benefits for fully reusable space transfer systems.
  - Cost of drop tanks may be more than cost to deliver additional propellant needed to recover tanks
  - Benefits indicated if we can separate functions of delivering hardware and delivering propellant. There are numerous launch system concepts for low-cost delivery of propellant to orbit; not applicable to delivering hardware.

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Mission / System Requirements Database

Gordon Woodcock

Agenda

Development Plan
Accomplishments Since Dec. 89
System Description
Complete Requirement Example
Reduced Graphics Example
Text Only Example

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STCAEM Requirements Database
System Description

Single User System

- Mac II
  - 1 Mb RAM
  - 40 Mb Hard Disk
  - 13 inch color monitor

- ACIUS 4th Dimension DBMS
  - Version 2.0
  - Customized User Interface

- 158 Entries
  - 3 Formats
    - Complete Requirement
    - Reduced Graphic
    - Text Only

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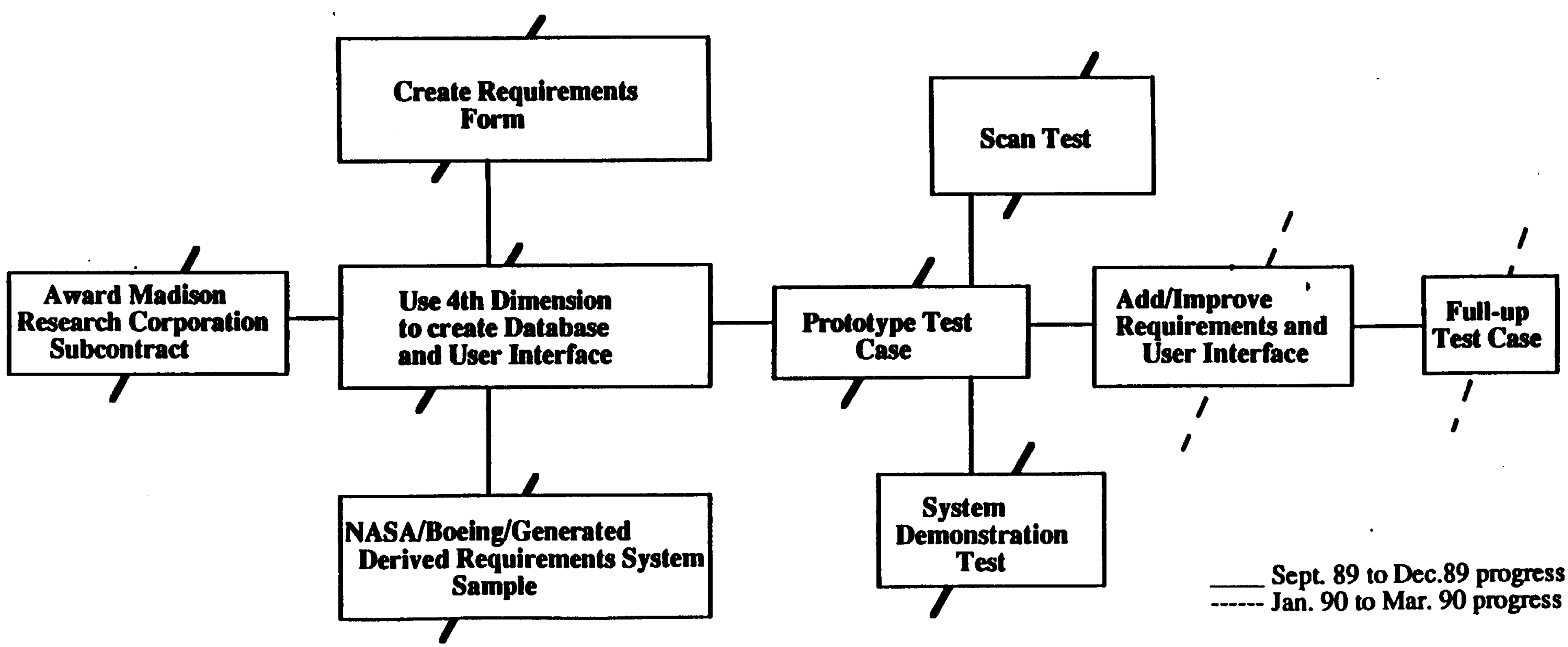
STCAEM Requirements Database Development Plan

Objectives

- Record and organize all requirements in text and graphics
- User-friendly reference, recall and manipulative tool



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Database Accomplishments Since Dec. 1989

- Upgraded from Mac SE to Mac II
- Improved User Interface
- Developed Users Manual
- Added Significant Number of Requirements
- Created Graphics Capability
- Improved Output Format

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STCAEM Requirements - Complete

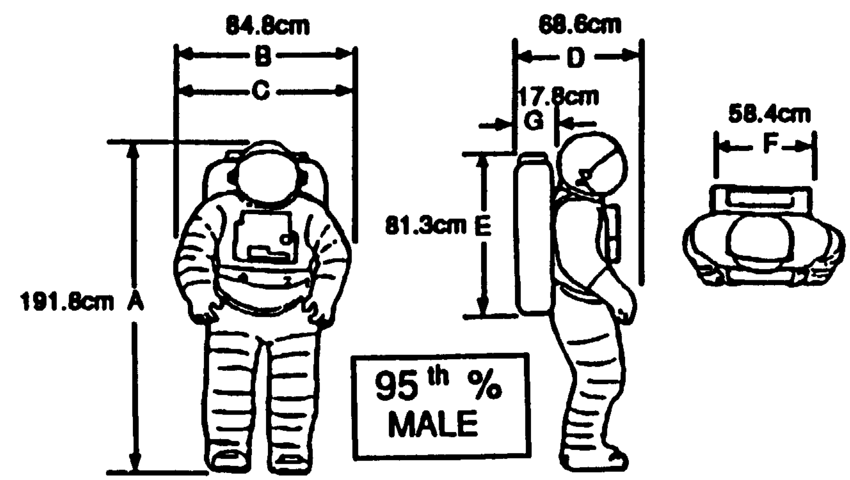
System	MTV	**Date**	01/05/90	**Reference** NASA-STD-3000, Vol. 1, Sect. 14.5.2.5
Subsystem	Transit Habitat	**Type**	Mission	Stephen Capps
Component	Circulation Corridors	**Origin**	NASA	(205) 461-3919

Requirement

Two (2) astronauts able to pass through major circulation paths while wearing spacesuits. (1.37 m x 1.92 m minimum).

Rationale

- Allows 2 astronauts to make repairs efficiently in the event of depressurization.
- NASA-STD-3000, Vol. 1, Sect. 14.5.2.5 states that "bottleneck impacts on crew productivity should be considered before EVA passageways are designed for one crew member."



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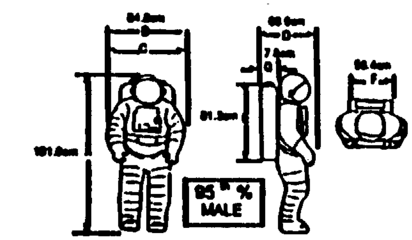
STCAEM Requirements Reduced Graphics

Date	Type	Orgin	System	Subsystem	Component
1/5/90	Mission	NASA	MTV	Transit Habitat	Circulation Corridors

Requirement

Two (2) astronauts able to pass through major circulation paths while wearing spacesuits. (1.37 m x 1.92 m minimum).



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Reference

NASA-STD-3000, Vol. 1, Sect. 14.5.2.5

Stephen Capps
(205) 461-3919

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STCAEM Requirements - Text Only

MTV — Transit Habitat — 1/5/90

Two (2) astronauts able to pass through major circulation paths while wearing spacesuits. (1.37 m x 1.92 m minimum).

- Allows 2 astronauts to make repairs efficiently in the event of depressurization.
- NASA-STD-3000, Vol. 1, Sect. 14.5.2.5 states that "bottleneck impacts on crew productivity should be considered before EVA passageways are designed for one crew member."

MTV — Transit Habitat — 1/5/90

Crew visibility during all maneuvers (docking/rendezvous).

NASA-STD-3000, Vol. 1, Section 8.11.3
- Windows are required for proximity operations.

MTV — Transit Habitat — 1/5/90

There shall be 2 means of egress from each module for emergency escapes.

NASA-STD-3000, Vol. 4, Sect. 8.7.3.4a

MTV — Transit Habitat — 1/5/90

Crew volumes contiguous

- Minimize repressurizations.
- Crew can inspect any crew volume at any time.

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Mission Analysis

ADVANCED
CIVIL
SPACE
SYSTEMS

Agenda

- Updated Trajectory Data
  - Mission dates and trajectories
  - Venus swingby
  - Orbit lighting
  - Communications
  - 2010 Mission

Mars Trajectory Data

The facing page summarizes trajectory data for most of the Mars mission profiles of interest. These data presume cryogenic/aerobraking high-thrust propulsion. Mars capture is defined as periapsis-to-periapsis transfer into a 250 km by 1 sol (24.6 hr) orbit. These trajectories do not apply to low-thrust systems.

For high-thrust nuclear propulsion, the trajectories are being re-optimized. Cryogenic/aerobraking profiles must pay all of the line-of-apsides misalignment penalty on Mars departure since aerocapture produces a periapsis-to-periapsis transfer. For a propulsive capture, the penalty may be divided between arrival and departure. Since the magnitude of the penalty is quite non-linear, a significant delta V savings is thereby obtained.

Mars Trajectory Data

STCAEM/pab/3/19/90

Opportunity		Earth Dep C3	Mars Arrival Vhp	Mars Arrival C3	Mars Departure Δ V	Earth Arrival Vhp	Earth Arrival C3	Mars Orbit Inc. (Deg)	Earth launch DLA	Mars Arr. LVI
2010 Opposition	12/01/10 -10/26/11, 11/25/11- 8/31/12	28.69	4.93	24.3	2.32	6.99	48.9	30	4.19	-28.63
2010 Conjunction	10/26/9 -10/31/10, 8/27/11 - 7/15/12	11.14	3.26	10.6	2.73	3.80	14.4	45	32.82	9.48
2013 Opposition	11/22/13 - 8/6/14, 10/5/14 - 8/14/15	13.08	4.10	16.8	3.54	4.53	20.5	30	20.42	25.32
2013 Conjunction	12/3/13 - 9/23/14, 9/28/15 - 9/6/16	9.58	3.15	9.92	2.61	4.47	20.0	30	23.73	31.13
2015 Level II Reference	5/23/15 - 4/22/16, 5/22/16 - 1/27/17	26.16	7.06	49.8	2.37	8.77	77.91	30	55.92	14.51
2015 Level II Alternate	10/15/15 - 7/16/16, 8/15/16 - 5/17/17	48.38	4.79	22.9	3.58	3.94	15.5	30	0.18	-12.22
2015 Conjunction	12/24/13 - 11/17/14, 12/14/15 -10/8/16	8.89	4.22	17.8	2.37	5.52	30.5	35	18.45	32.93
2015 L II Ref. + 50 day	5/23/15 - 4/22/16, 5/22/16 - 1/27/15	14.19	6.93	48.0	2.37	9.47	89.7	30	55.92	15.22
2016 Boeing Nominal	2/25/16 - 7/31/16, 8/31/16 - 5/51/17	10.34	6.82	46.5	4.30	7.14	51.0	30	-35.94	-1.69
2018 Opposition	3/27/17 - 3/10/18, 4/24/18 - 12/18/18	19.71	5.96	35.5	2.58	5.04	25.4	30	17.97	22.3
2018 Conjunction	5/12/18 -11/28/18, 5/31/20 - 11/27/20	7.86	2.97	8.8	3.41	4.02	16.2	45	-37.94	-7.81
2020 Opposition	6/4/20 - 12/11/20, 1/10/21 - 1/28/22	24.40	3.89	15.1	4.27	4.28	18.3	25	15.11	-9.69
2020 Conjunction	7/20/20 - 1/16/21, 8/09/22 - 1/16/23	13. 40	3.13	9.8	3.92	6.67	44.5	30	18.65	-3.40
2023 Opposition	11/11/21 - 9/17/22, 10/17/22 - 6/5/23	16.31	5.31	28.2	4.41	6.00	36.0	30	-60.02	-4.04
2023 Conjunction	9/8/22 - 4/16/23, 7/9/24 - 5/5/25	19.03	3.18	10.1	2.66	2.86	8.81	30	50.73	23.71
2024 Opposition	9/20/23 - 7/4/24, 8/4/24 - 5/30/25	27.91	6.46	41.7	1.61	3.08	9.49	30	-19.58	-6.53
2024 Conjunction	10/17/24 - 6/24/25, 8/11/26 - 5/5/27	19.68	3.00	9.0	2.60	2.60	6.76	35	55.88	34.75

Venus Swingby Data

All of the opposition mission profiles employ Venus swingby to reduce mission delta V. Swingby data are presented on the facing page.

Venus Swingby Data

Opportunity	Outbound Venus miss distance / date (km/ Julian 245XXX.X)	Inbound Venus miss distance / date (km/Julian 245XXX.X)	Δ V difference leg to leg km/sec
2010 Opposition	7761 / 198.43		-0.3
2013 Opposition		12003 / 168.2	-7.2
2015 Level II Reference	13276 / 150.4		350.0
2015 Level II Alternate		16439 / 125.1	-1036.3
2015 L II Ref. + 50 day	13276 / 150.4		350.0
2016 Boeing Nominal		16361 / 191.0	-97.0
2018 Opposition	10140 / 166.47		6.7
2020 Opposition		8794 / 209.26	6.8
2023 Opposition	16006 / 140.19		-4.04
2024 Opposition	11553 / 151.5		228.3

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Trajectory Information for Mission Opportunities

Opportunity	C3 Earth Departure	Vhp Mars Arrival	C3 Mars Departure	Vhp Earth Arrival	Periapsis Altitude (km)	Apoapsis Radius (km)	Periapsis Radius (km)	Eccentricity	Semi-major Axis (km)
2010 Opposition	28.69	4.93	16.07	6.99	250	37182.86	3647	0.82	20414.93
2010 Conjunction	11.14	3.26	7.04	3.72					
2013 Opposition	13.08	4.10	37.68	4.53					
2013 Conjunction	9.58	3.15	6.88	4.47					
2015 Level II Reference	20.16	7.02	29.97	8.77					
2015 Level II Alternate	48.38	4.79	33.28	3.94					
2015 Conjunction	8.89	4.22	5.42	5.52	500	36932.86	3897	0.81	
2015 L II Ref. + 50 day	14.19	6.93	17.97	9.47	250	37182.86	3647	0.82	
2016 Boeing Nominal	10.34	6.82	39.70	7.14					
2018 Opposition	18.45	6.83	6.33	3.26					
2018 Conjunction	7.86	2.97	12.28	4.02					
2020 Opposition	31.49	3.87	36.89	7.71					
2020 Conjunction	13.40	3.13	20.03	6.67					
2023 Opposition	23.29	6.53	43.27	5.29					
2023 Conjunction	19.03	3.18	9.31	2.86					
2024 Opposition	27.91	6.46	9.02	3.08					
2025 Conjunction	19.68	3.00	7.81	4.79					

Orbit Insertion Lighting Angle, Latitude, and Approach Turning Angle for 2010 to 2025 Mars Mission Opportunities

All of the selected mission profiles have a daylight periapsis; this is necessary to ensure daylight conditions for landing since the landing will occur near periapsis. Also, we have tried to keep periapsis latitude within ± 30° of the Mars equator.

Orbit Insertion Lighting Angle, Latitude, and Approach Turning Angle for 2010 to 2025 Mars Mission Opportunities

Opportunity	Periapsis Lighting Angle (°)	Periapsis Latitude (°)	Approach Turning Angle (°)
2010 Opposition	54.19	1.21 N	70.96
2010 Conjunction	42.20	42.51S	58.30
2013 Opposition	21.94	24.33 S	65.70
2013 Conjunction	55.01	16.40 S	57.22
2015 Level II Reference	22.57	28.22 S	78.78
2015 Level II Alternate	35.45	29.97 S	70.19
2015 Conjunction	67.75	22.12 S	67.58
2015 L II Ref. + 50 day	23.52	29.47 S	78.67
2016 Boeing Nominal	11.15	28.88 S	78.37
2018 Opposition	36.54	24.21 S	78.39
2018 Conjunction	50.52	29.47 S	55.16
2020 Opposition	13.50	15.97 S	63.94
2020 Conjunction	10.67	22.59 S	57.01
2023 Opposition	66.41	32.45 N	77.90
2023 Conjunction	10.61	1.99 N	57.50
2024 Opposition	68.29	26.75 S	77.31
2025 Conjunction	15.32	22.06 S	55.55

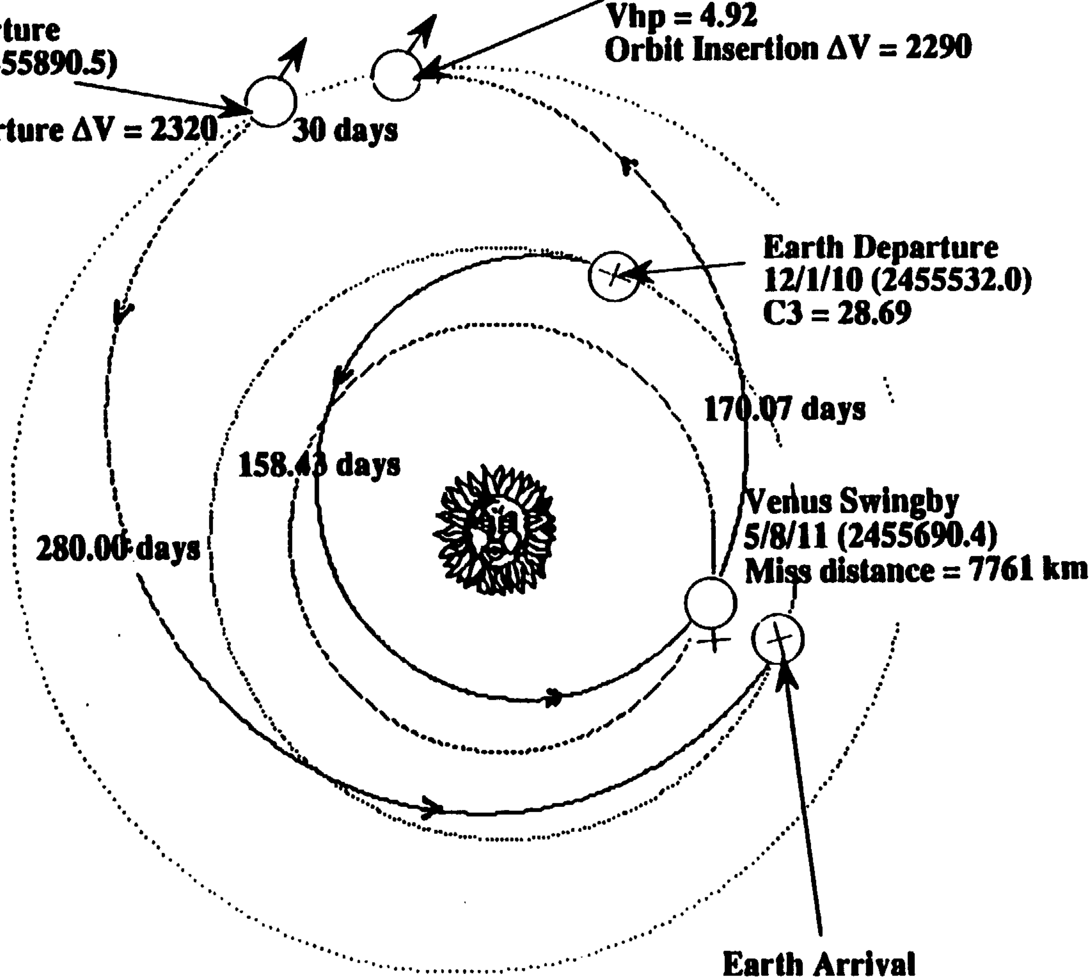
Data generated by the PLANET program, property of the Boeing Company.

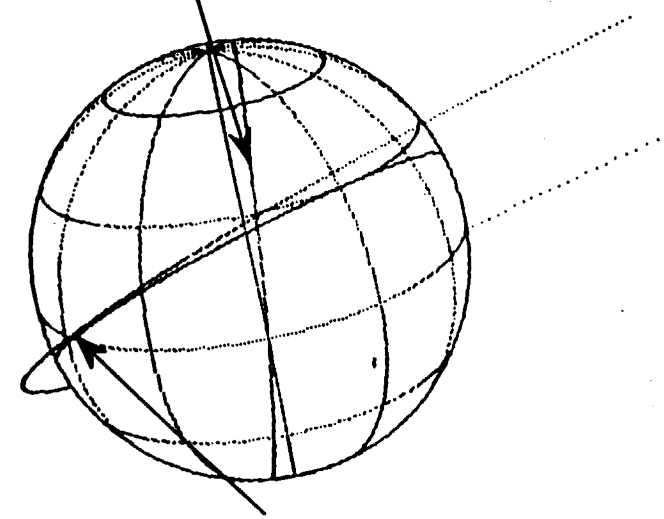
2010 Opposition Class Mission Trajectories

The 2010 mission opportunity trajectory design is illustrated here. This profile is of interest for an early Mars mission because its energies are relatively low, leading to the possibility of an early Mars mission with lunar transfer vehicle propulsion systems. The Mars arrival C3 is about 25, alleviating heating concerns. The periapsis latitude is near the equator, so that a low L/D lander could be used for a first mission. The main disadvantage of this opportunity is that the trip time is relatively long. It takes almost a year to get to Mars with an outbound Venus swingby, and over nine months to get back to Earth after a 30-day stay at Mars. This seems to be characteristic of the low-energy opportunities. The low-energy Venus swingby repeats about every 6.5 years. The 2004 and 2017 low-energy opportunities have similar long trip times.

For the 2015/16 opportunity, we found an inbound Venus swingby 200 days shorter (at 434 days), but with higher energies. Depending on the relative importance of trip time and energy, opportunities for a first Mars mission appear to be 2010 or 2015/16.

2010 Opposition Class Mission Trajectories





Total trip time = 638.50 days

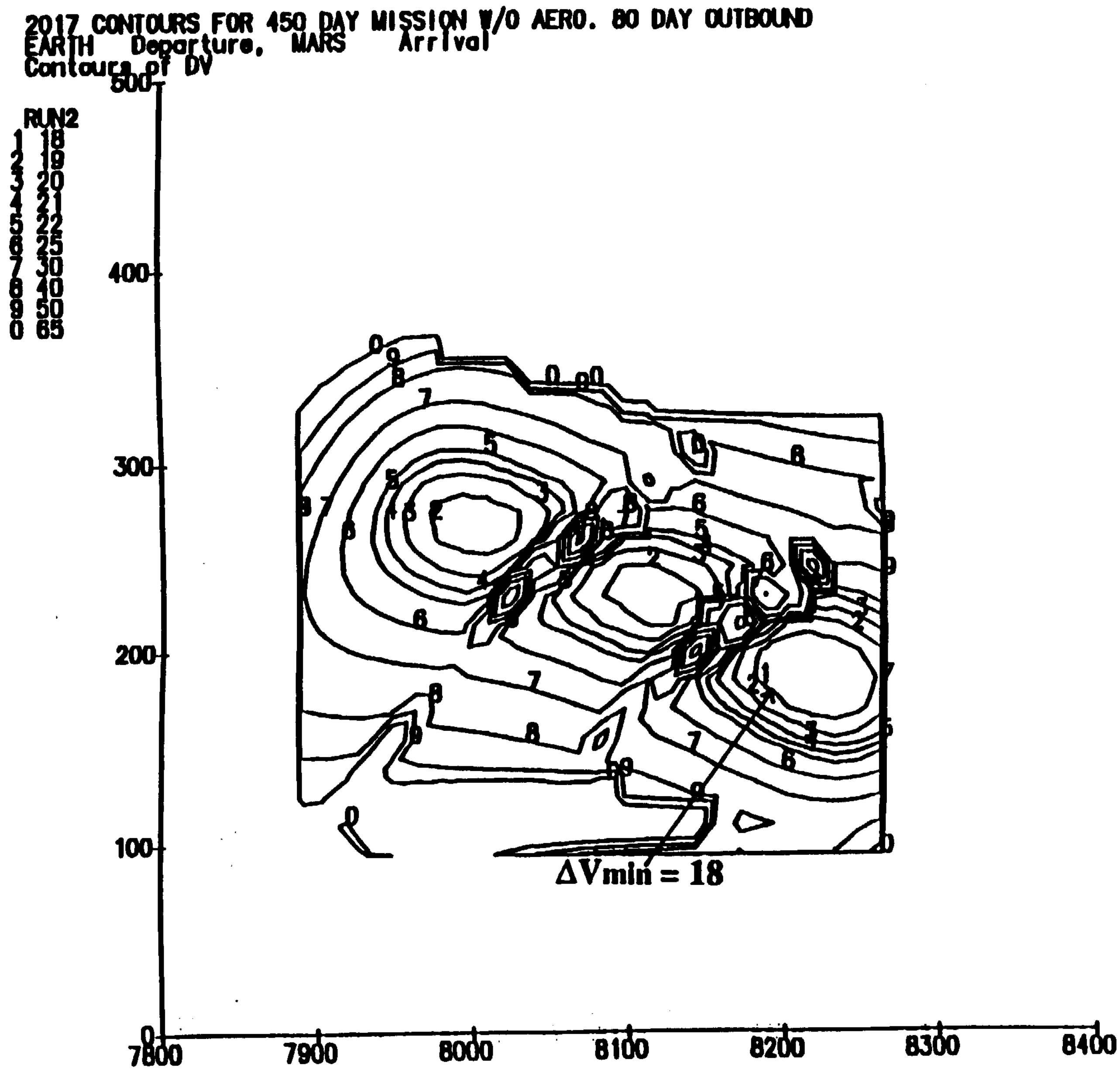
2017 Opportunity, 450-day Mission, No Aerobraking

The facing page shows a graphic technique we are developing to quickly find optimum opportunities. In order to display contours of mission energy on a 2-dimensional plot, the mission is constrained by fixing the trip time and the Mars stay time. This leaves two free parameters, the date of departure on the abscissa (Julian date, 245XXXX), and the duration of the Earth-Mars leg in days on the ordinate. The graph shown is for an all-propulsive mission without Venus swingby.

Boeing-funded work now in progress is aimed at adding an automated swingby selection routine, so that regions where a swingby results in least delta V can be included on these graphs.

2017 Opportunity ,450 Day Mission
80 Days Outbound, No Aerobraking

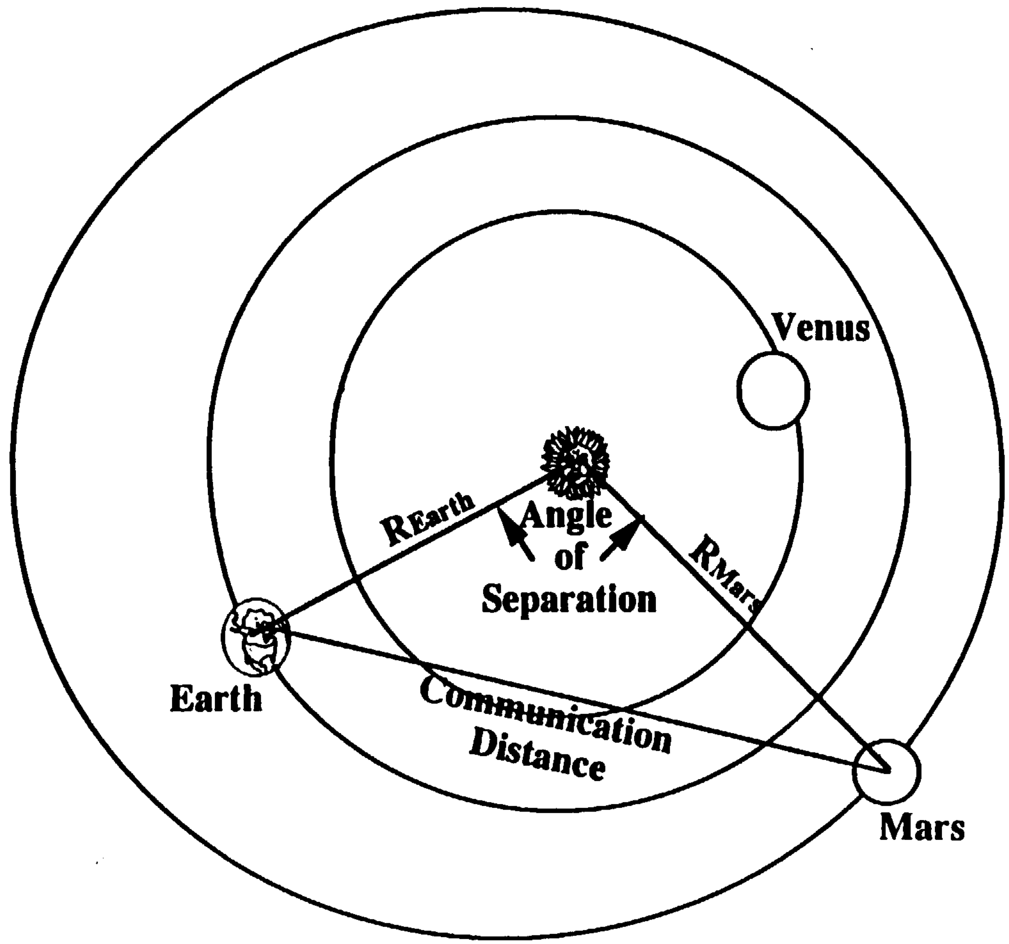
STCAEM/pth/30Jan90



Coordinate Definition

This figure illustrates the coordinates for definition of the Earth-Mars communications distance at arrival.

Coordinate Definition



Communications Coordinates at Mars

Communications distances and separation angles at Mars arrival are shown here for the various Mars opportunities.

Communications Coordinates at Mars

Opportunity	Communication Distance at Mars (x 10^6 km)	Heliocentric Coordinates at Mars Arrival		
		R_{Earth} (x 10^6 km)	R_{Mars} (x 10^6 km)	Separation Angle (rad)
2010 Opposition	291	151	232	1.46
2010 Conjunction	348	128	222	0.71
2013 Opposition	184	152	221	2.17
2013 Conjunction	224	150	213	1.85
2015 Level II Reference	95	150	234	2.90
2015 Level II Alternate	96	152	217	2.75
2015 Conjunction	271	149	207	1.43
2015 L II Ref. + 50 day	94	150	234	2.90
2016 Boeing Nominal	108	152	215	1.57
2018 Opposition	218	148	236	2.02
2018 Conjunction	148	148	212	2.37
2020 Opposition	97	147	220	2.78
2020 Conjunction	155	147	229	2.41
2023 Opposition	84	148	225	2.96
2023 Conjunction	240	150	248	1.93
2024 Opposition	258	152	210	1.58
2025 Conjunction	281	152	246	1.63

Data generated by the PLANET program, property of the Boeing Company

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GN&C, Aeroheating, Mars Landing

Agenda

- GN&C
  - Mars atmosphere
  - Mars aeroentry maneuvers
- Aeroheating
- Landing
  - Landing scheme
  - Crossrange & ideal landing ΔVs
  - Aerobrake shape
  - Results

Aerocapture GN&C Analyses

Steps in development of aerocapture analyses are described here. As indicated by the heavy check marks on the chart, the first three types of analysis are complete or nearly complete for the present effort. We are presently in the early phases of developing guidance schemes. We do not plan to perform six-degree-of-freedom simulations on the present study, as the expense would result in an unbalanced application of available resources. We believe that the 3-1/2 DOF simulations (including finite roll rates) are adequate to establish aerocapture GN&C feasibility.

Aerocapture GN&C Analyses

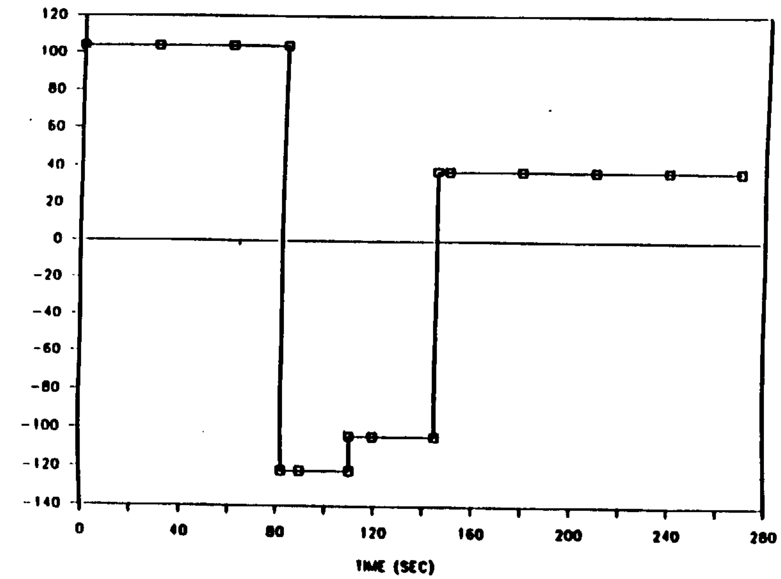
Analysis Type		Results Obtained
• Closed-form zero lift approximation; fixed exponential atmosphere	√	Depth of penetration versus ballistic coefficient and entry velocity
• Fixed-lift integrated trajectories; 2-DOF; fixed tabulated atmospheres	√	Corridor height and g level vs. available L/D and entry velocities; entry conditions
• Modulated lift integrated trajectories; 3-DOF or 3-1/2 DOF; fixed tabulated atmosphere	√	Trajectory designs for aerocapture, considering vehicle lift modulation capability and rates
• Modulated lift integrated trajectories; 3-DOF or 3-1/2 DOF; variable atmosphere	√	Development of guidance schemes and laws; assessment of errors induced by atmoshpere unpredictability
• 6-DOF integrated trajectories with simulation of vehicle flight control system; variable atmosphere		Accurate assessment of vehicle capabilities for aerocapture; detailed design requirements for aerobrakes and flight control systems

Guidance, Navigation & Control

This shows another example of the trajectory design, here using the OPTIC code. We are using two different GN&C codes to cross-check results. OPTIC, developed by Boeing-Seattle, optimizes with constraints. The other, AEROPASS, developed by Boeing-Huntsville, optimizes switch points and exercizes guidance laws. Constraints must be represented by penalty functions with this routine.

Guidance Navigation & Control

LIFT MODULATION VIA BANK ANGLE MOD.

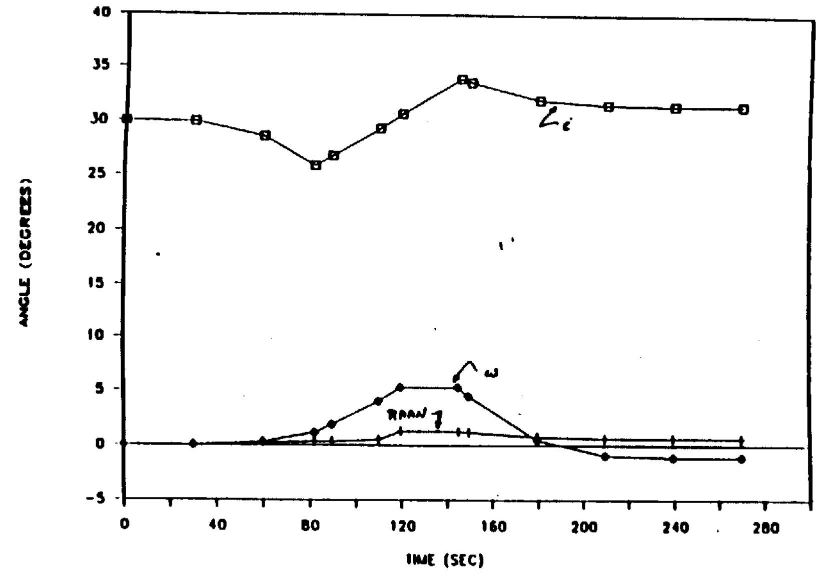


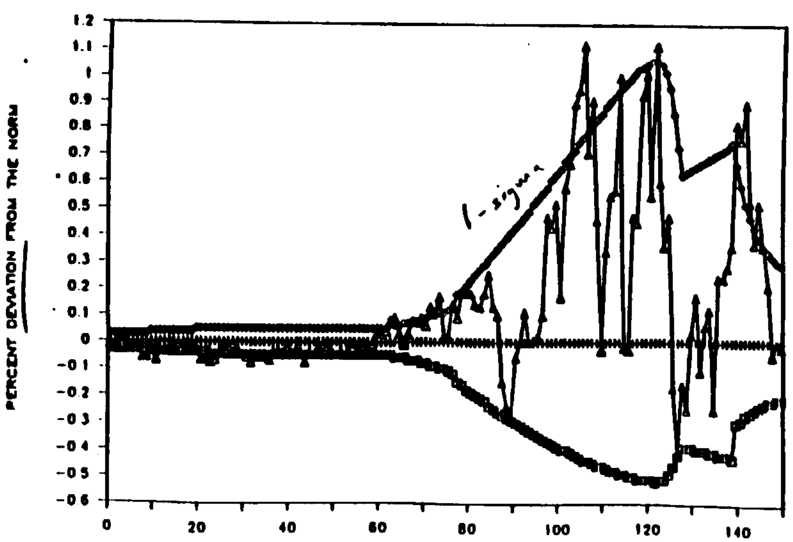
< Guidance control through the Mars atmosphere is done by controlling bank angle in a "slalom course" motion

>

This is the resultant inclination,right ascension angle and argument of periapsis (position with respect to the planet) for the "slalom course" run

INC,RAAN, & ARGPER PROFILES





< comparison of the atmospheric deviation obtained with the MARSGRAM and Optic codes

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Guidance Schemes for Aerocapture

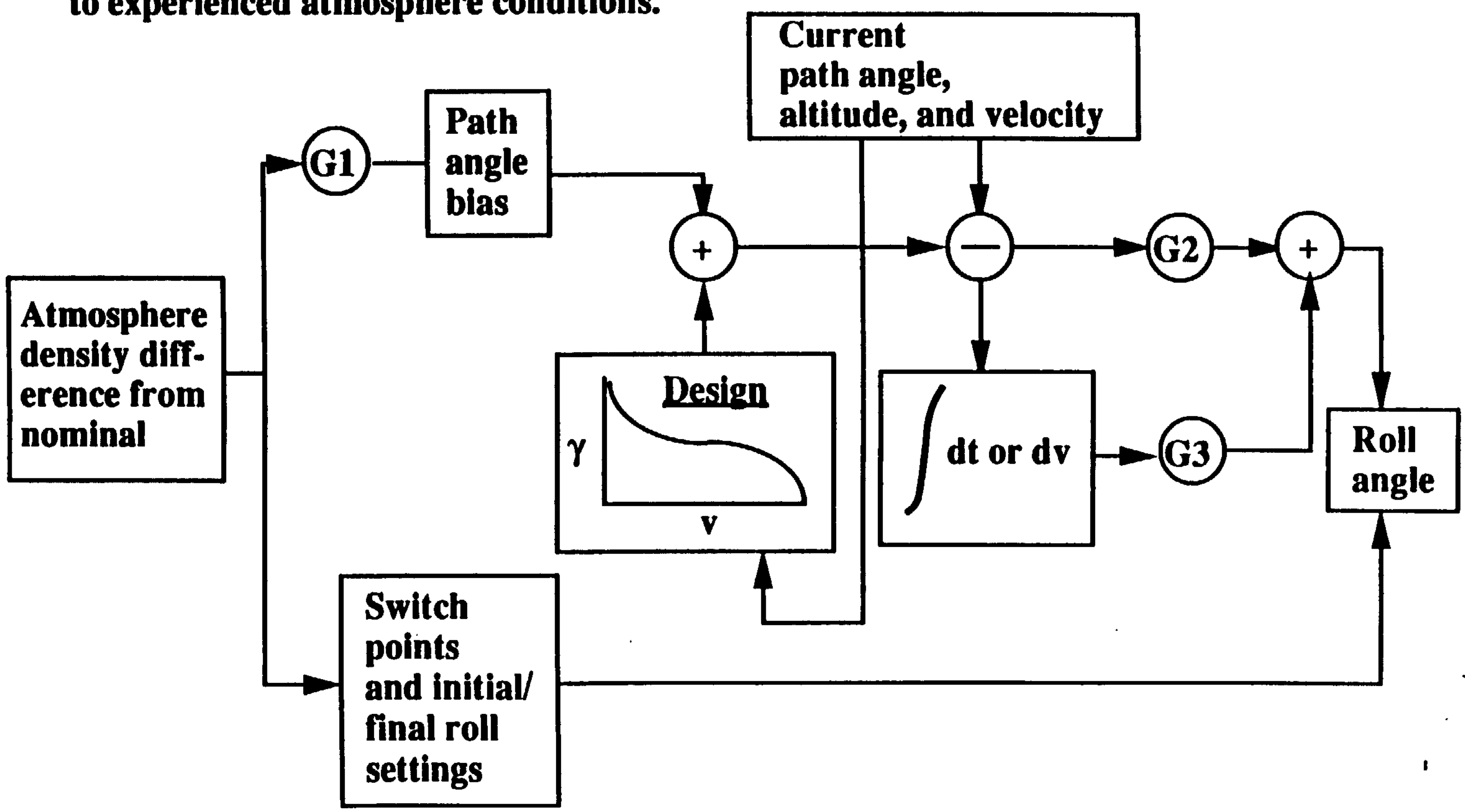
The facing page identifies the requirements for aerocapture GN&C, summarizes two approaches to adaptive guidance, and illustrates a gain-scheduling scheme presently under investigation. We also plan to investigate the real-time re-optimization scheme.

ADVANCED CIVIL SPACE SYSTEMS

Guidance Schemes for Aerocapture

BOEING

- Requirements - minimize changes in inclination and line of nodes. Attain desired line of apsides and apoapsis altitude.
- Maximum performance - redesign trajectory optimization and constraints every few seconds, but requires very high computer performance.
- Less performance but may be adequate - use an adaptive gain scheduling scheme that adjusts to experienced atmosphere conditions.

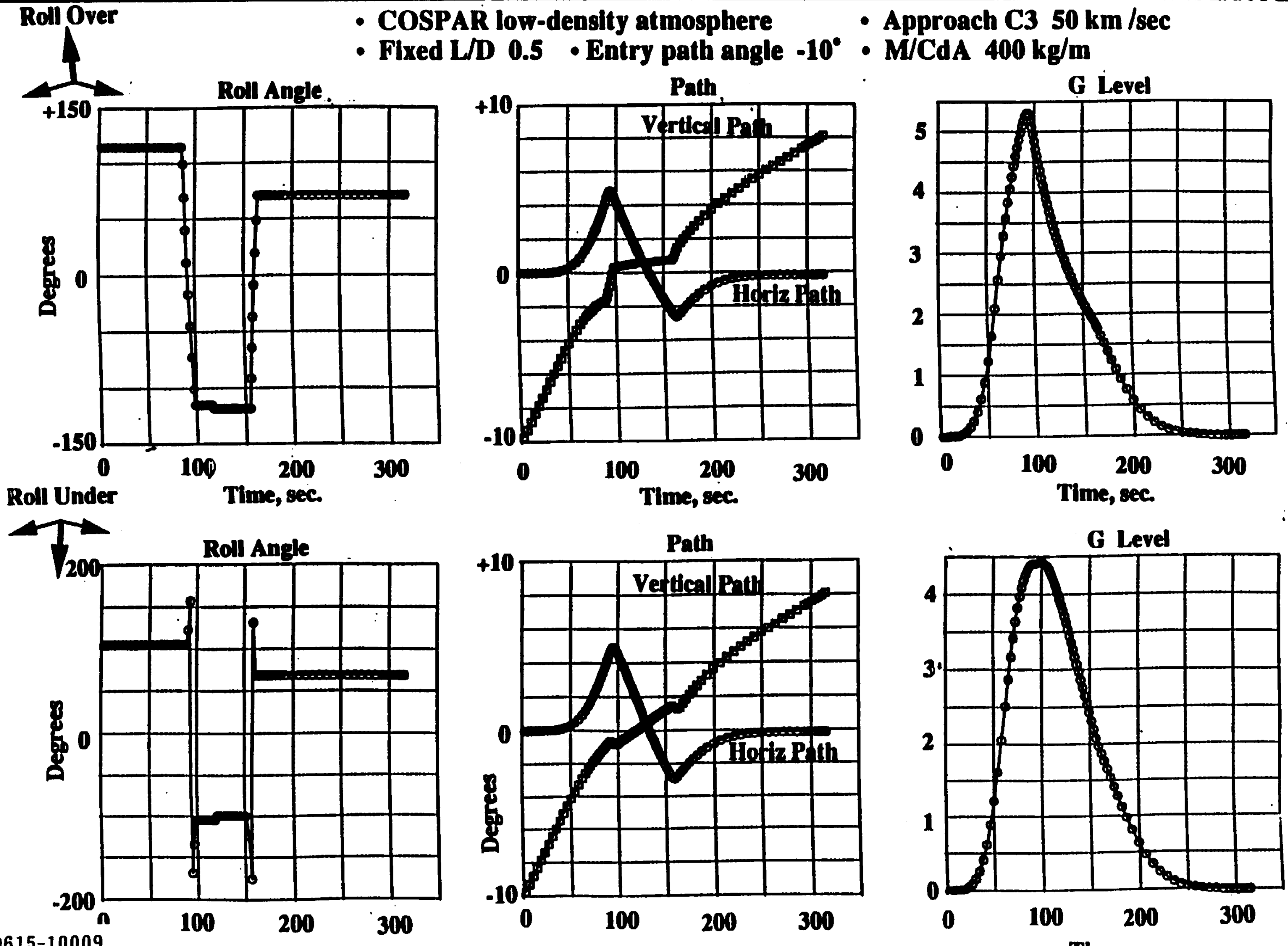


Mars Aerocapture Trajectory - Finite Roll Rate

This figure shows the effects of finite roll rates on the trajectory design. In going from left to right and back, the lift vector may be rolled over the top, or under. Because this perturbs the vertical path, the rest of the trajectory design must compensate by going a little deeper (roll over) or a little less deep (roll under). These results show that the effect on the vertical path is less with the roll under, and that the maximum deceleration is less, leading to a clear preference for "roll under".

Mars Aerocapture Trajectory - Finite Roll Rate

- COSPAR low-density atmosphere
- Fixed L/D 0.5
- Entry path angle -10°
- Approach C3 50 km /sec
- M/CdA 400 kg/m

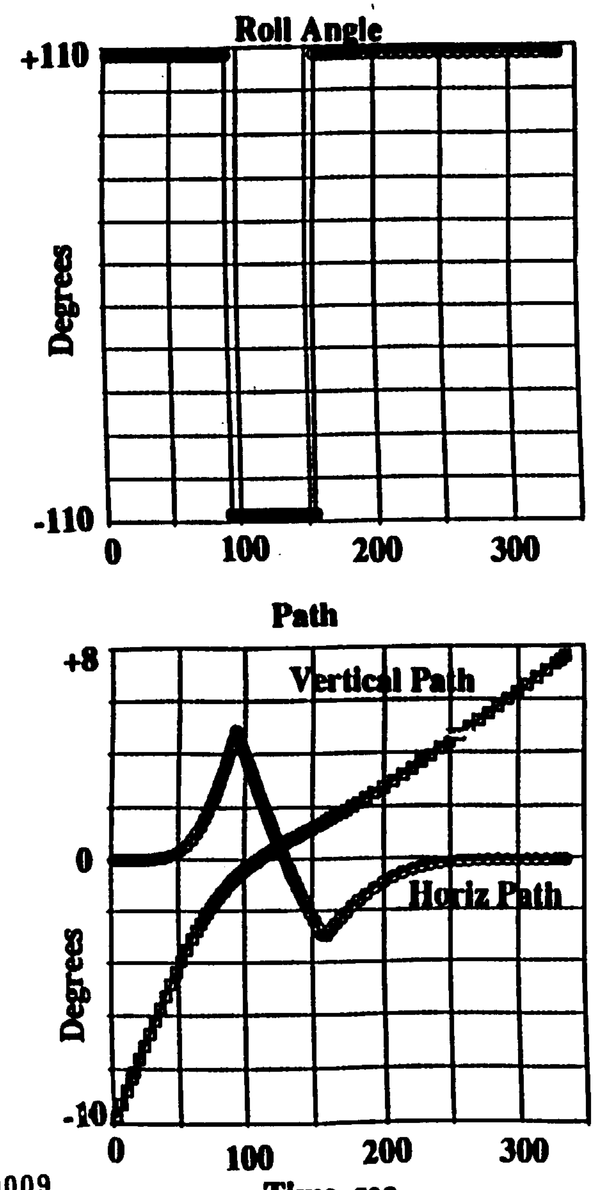


Mars Aerocapture Trajectory Design Approach

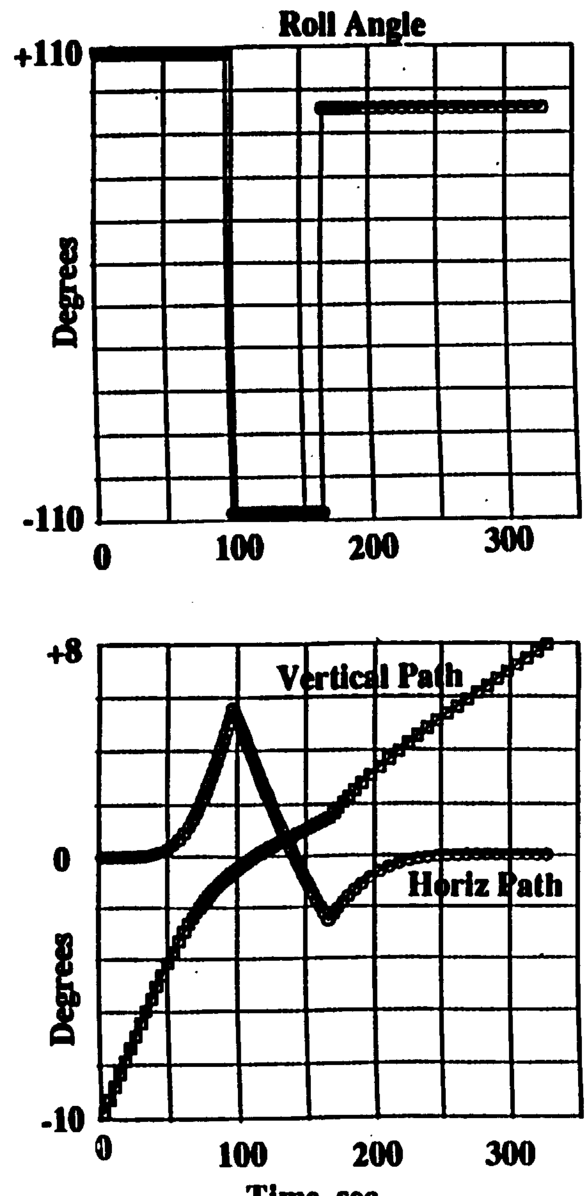
The trajectory design approach is tailored to a roll-only control scheme. Excess lift is dissipated by veering the trajectory to the left and to the right in a dog-leg or "slalom" maneuver. The illustration on the left shows a nominal symmetric design, with roll angles of 107° (0° is straight up). This applies a net vertical lift coefficient of about -0.15. The asymmetric design on the right enables control of the line of apsides, so that the range of atmospheres represented by the COSPAR low and high density atmospheres can be navigated, from the same entry conditions, to realize the same capture orbit, within reasonable delta V budget for post-exit correction. For the low-density atmosphere, the roll angle is greater during penetration than during exit. For the high-density atmosphere, the reverse is true.

Mars Aerocapture Trajectory Design Approach

- COSPAR low-density atmosphere
- Fixed L/D 0.5
- Entry path angle -10°
- Approach C3 50 km /sec
- M/CdA 400 kg/m



Basic design is symmetric

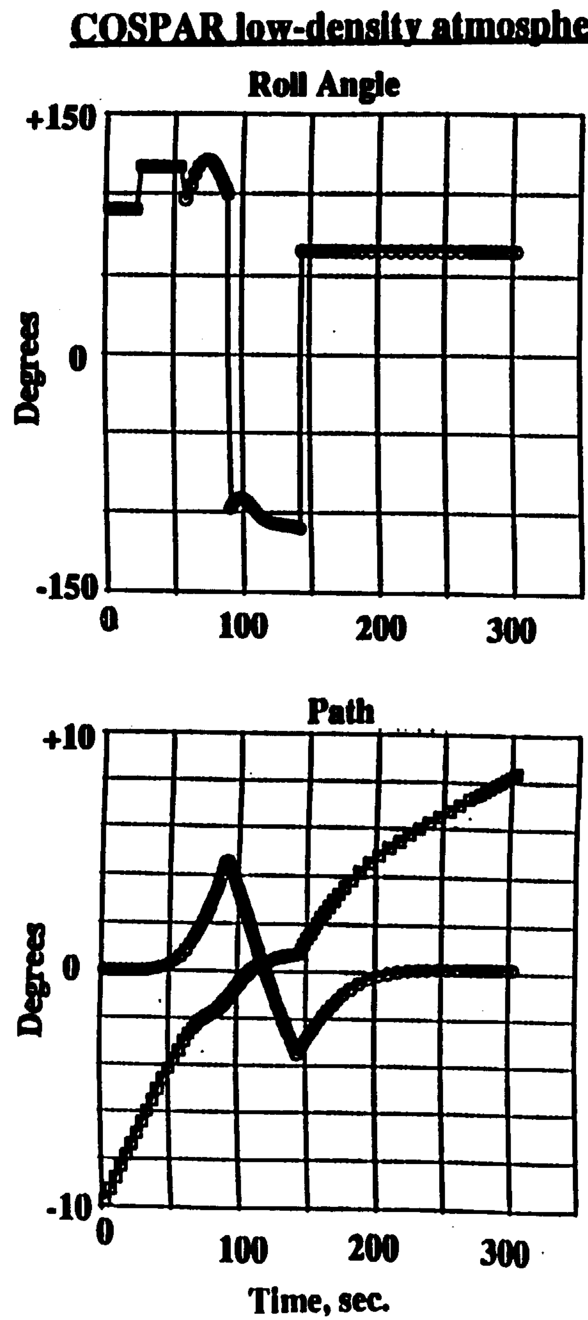


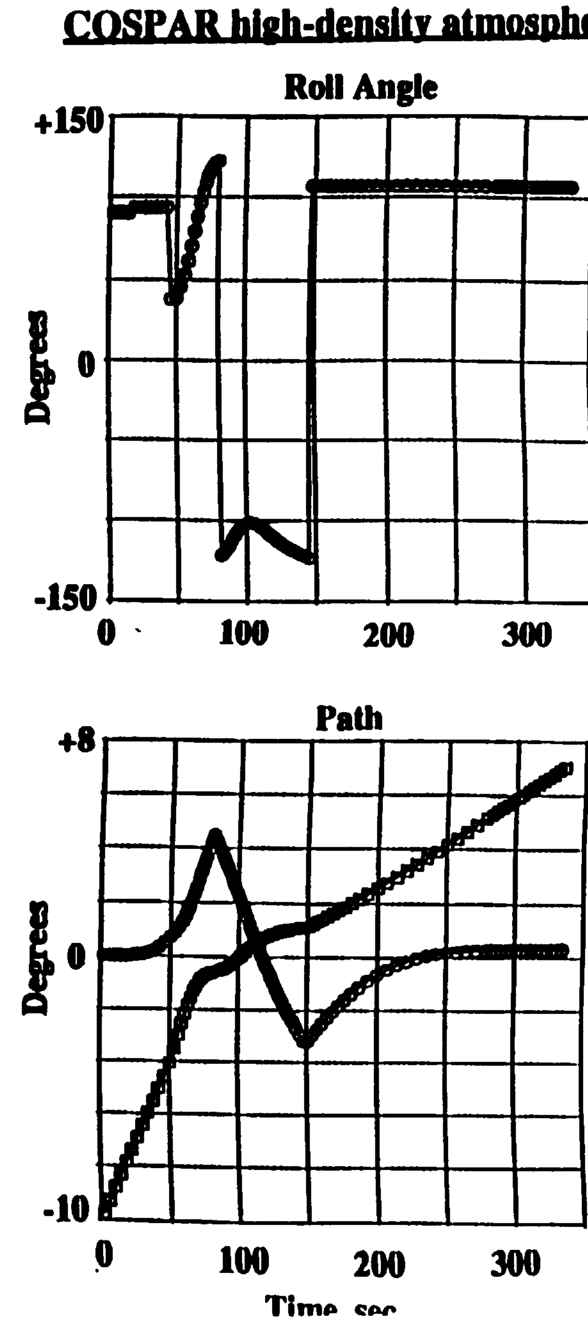
Asymmetric roll program enables line-of-apsides control

Mars Aerocapture - Guided Trajectory Examples

This shows preliminary results of trying the gain-scheduling scheme. Gains have not been optimized. The same entry conditions were used with the high and low-density atmospheres. The guidance scheme gave good results on all exit conditions except line of apsides, and fair results for that parameter.

Mars Aerocapture - Guided Trajectory Examples



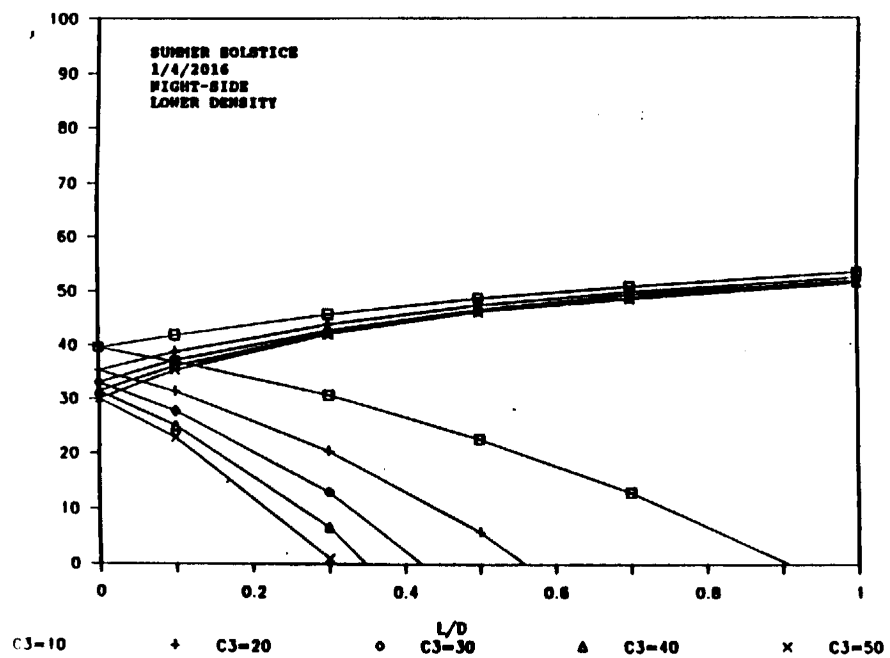


Atmosphere Entry Conditions for "Slalom Course" Maneuver

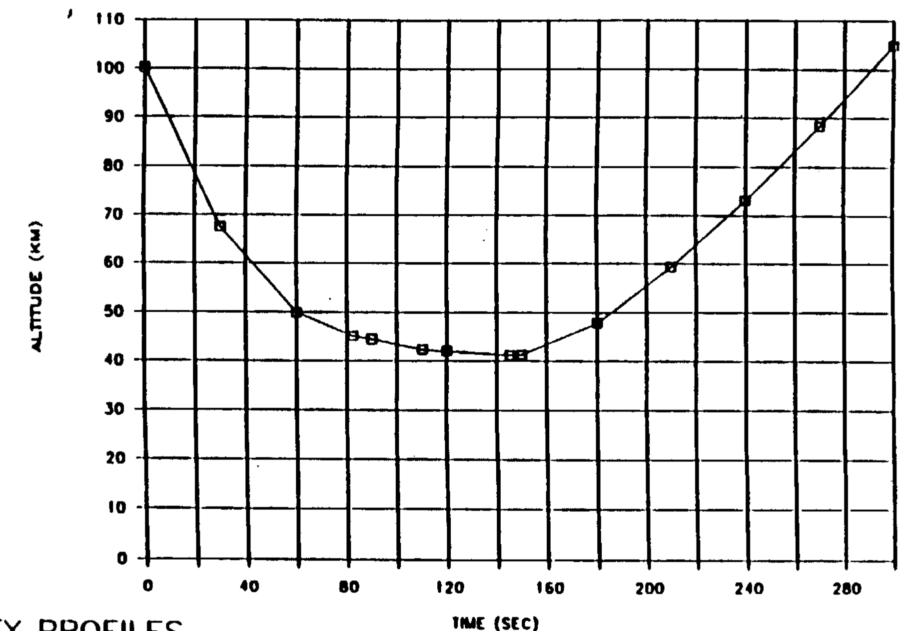
An additional display of the trajectory design is shown here. Corridor height parametrics are on the left. A typical trajectory profile for a relatively dense MARS-GRAM atmosphere is on the upper right. Typical MARS_GRAM atmosphere desnity predictions are shown on the lower right.

Atmospheric Entry Conditions for "Slalom Course" Maneuver

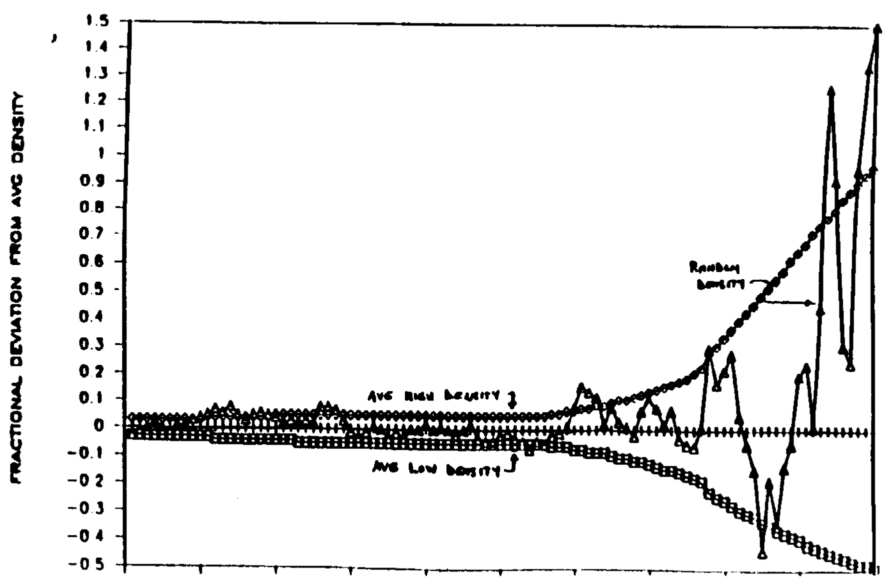
CORRIDOR PROFILE



TRAJECTORY PROFILE



DENSITY PROFILES

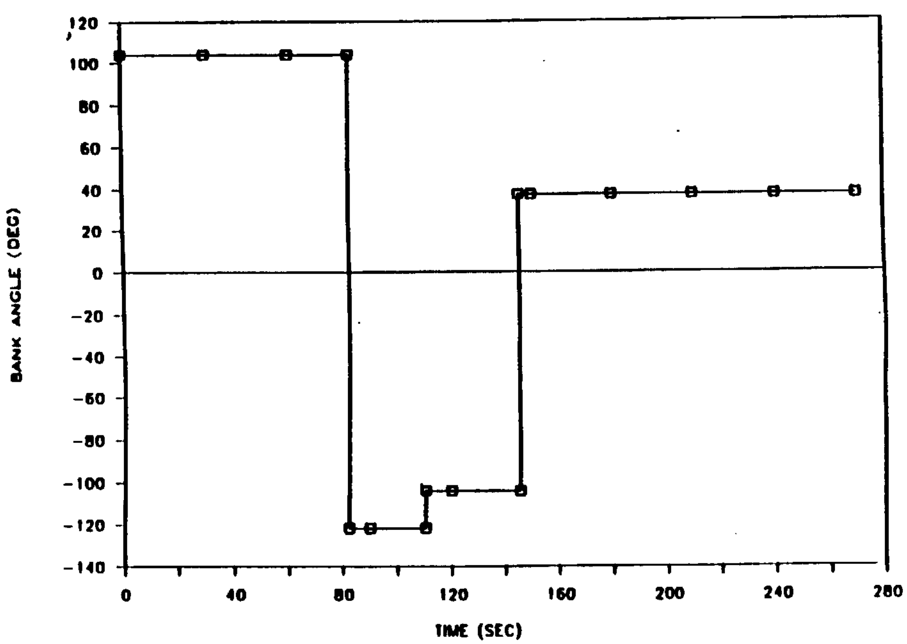


"Slalom Course" Maneuver Profiles

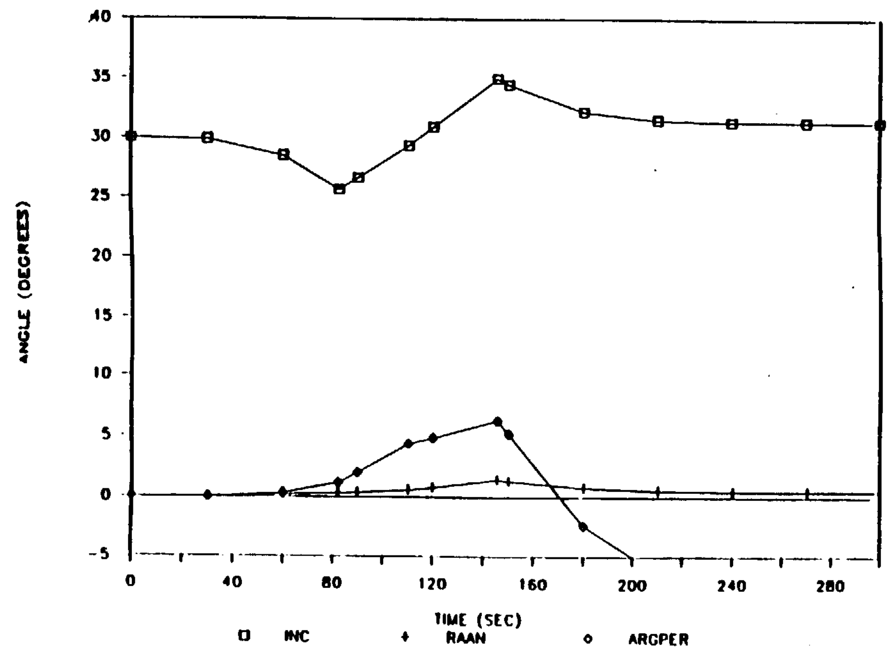
This simulation, with the OPTIC code, examined the effects on a trajectory design of typical atmosphere density variations predicted by MARS-GRAM. The most significant effects were a significant reduction in exit velocity and a large rotation of the line of apsides. No adaptive guidance was simulated. The result shows a clear need for adaptive guidance.

"Slalom Course" Maneuver Profiles

LIFT MODULATION VIA BANK ANGLE MOD.



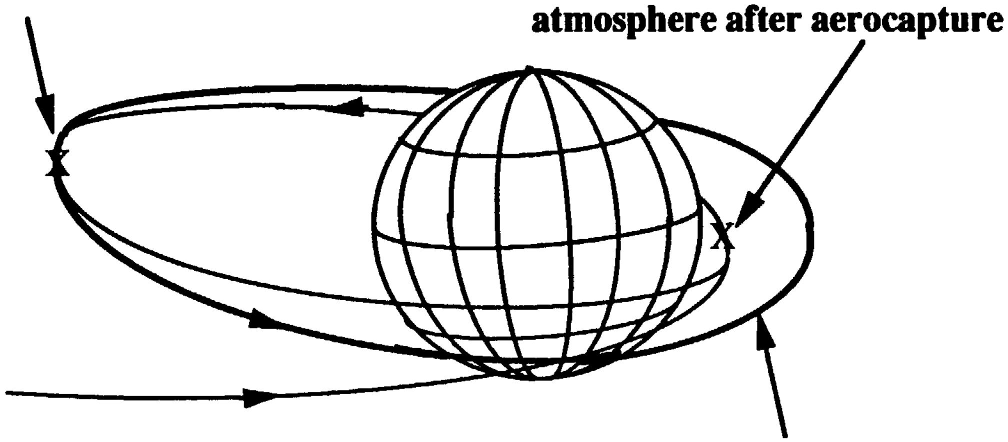
INC,RAAN, & ARGPER PROFILES

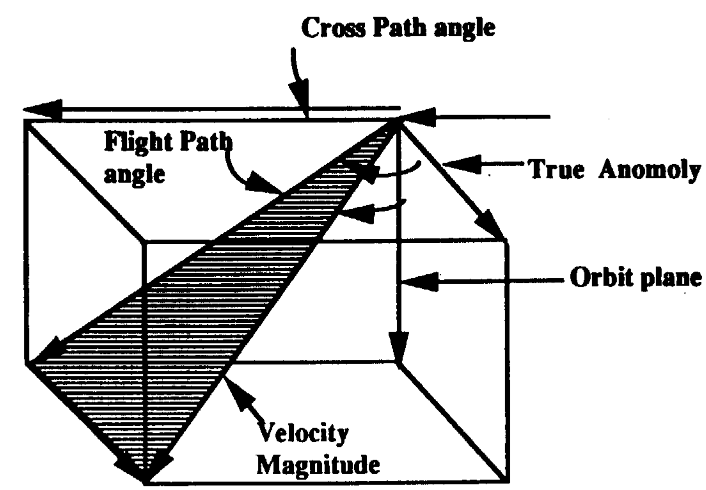


Orbit Correction Analysis

The next three pages illustrate a method of correcting exit conditions with two burns, and show some preliminary results of calculations of the delta V required for each burn as a function of the magnitude of exit errors.

Orbit Correction Analysis





- Burn at V1 will - align the apsides
  - set the apoapsis height to the desired final orbit height
- Burn at V2 will - raise periapsis height
  - correct inclination errors

- State vector errors are:
  - flightpath angle
  - crosspath angle
  - velocity magnitude
- Position Vectors errors are:
  - true anomaly
  - orbit plane

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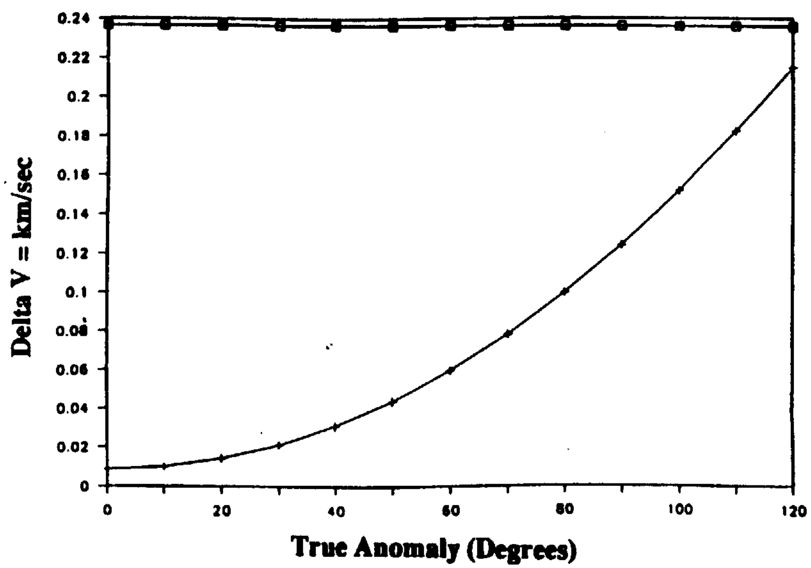
Effect of Burn Location on ΔV Penalties

Page 1

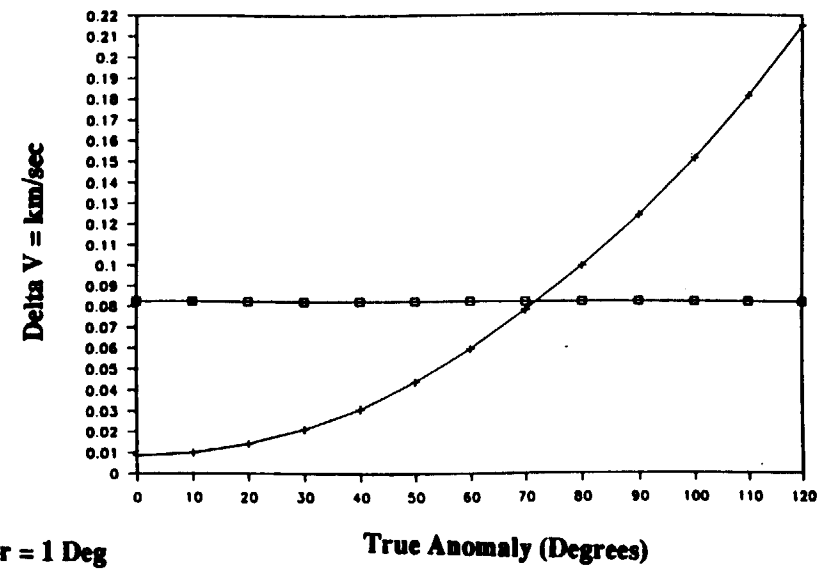
Velocity Magnitude Error = 0.95

DELTA V VS. TRUE ANOMALY



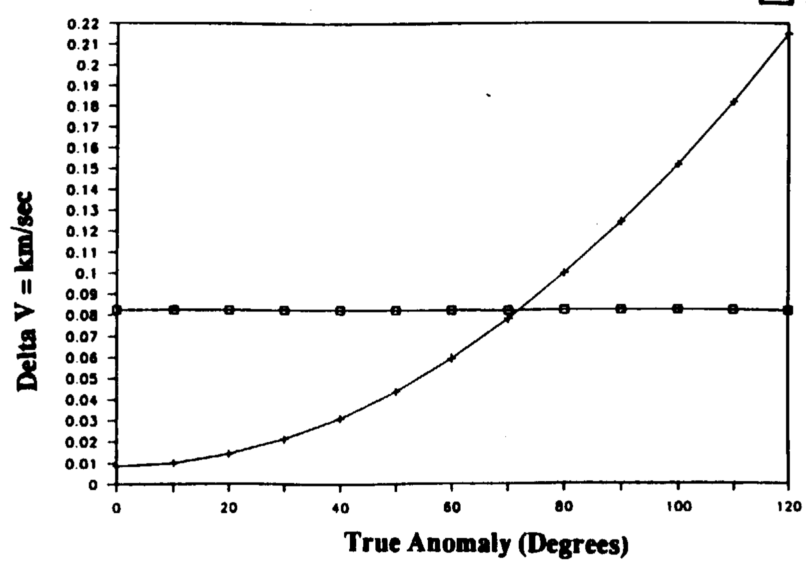
Flight Path Angle Error = 1 Deg

DELTA V VS. TRUE ANOMALY



Cross Path Angle Error = 1 Deg

DELTA V VS. TRUE ANOMALY



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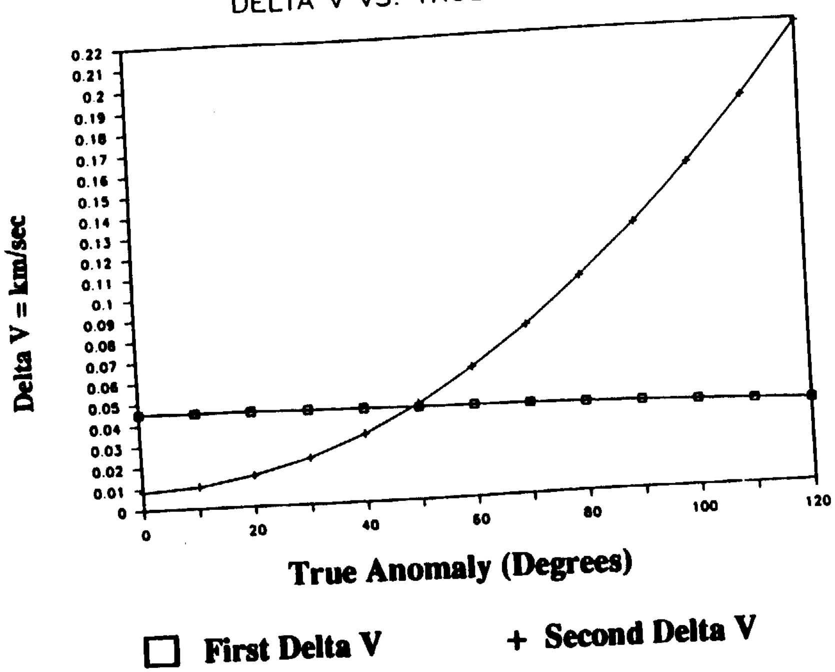
Effect of Burn Location on ΔV Penalties

Page 2

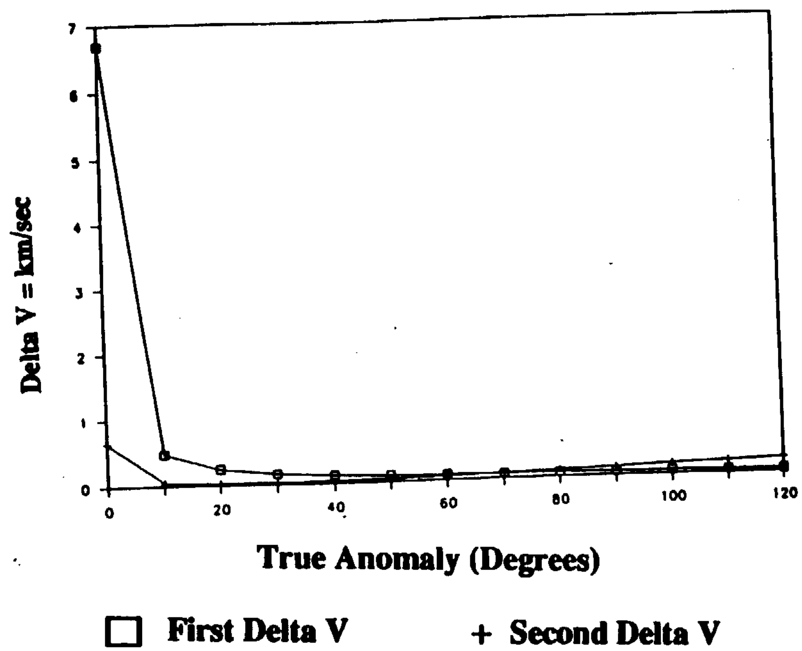
True Anomaly Error = 1 Deg

DELTA V VS. TRUE ANOMALY



Position Plane Error = 1 Deg

DELTA V VS. TRUE ANOMALY



(Aeroheating) Methods and Assumptions

Aeroheating estimation methods we are using are summarized here.

C 2

Aeroheating Methods and Assumptions

Stagnation Point Heating

- **Radiative (Park Method)**
  - Equilibrium (Stagnation Pressure > 0.1 atm)
  - Optically thick gas (absorptivity = 0.5)
  - Park Method reliable to within ± 30%

- **Convective**
  - Modified Fay-Riddell
  - Fully Catalytic

Distributed Heating(Continuum flow)

- **Radial Streamlines Assumed**

- **Radiation**
  - Approximate shock shape used.
  - Averaged Normal Velocity Component is the used in the Park Method

- **Convective (Boundary Layer Analysis Program)**
  - Axisymmetric Analog
  - Pressure Distribution: Newtonian Impact Theory
  - Laminar Flow (Re transistion = 2×10^6)

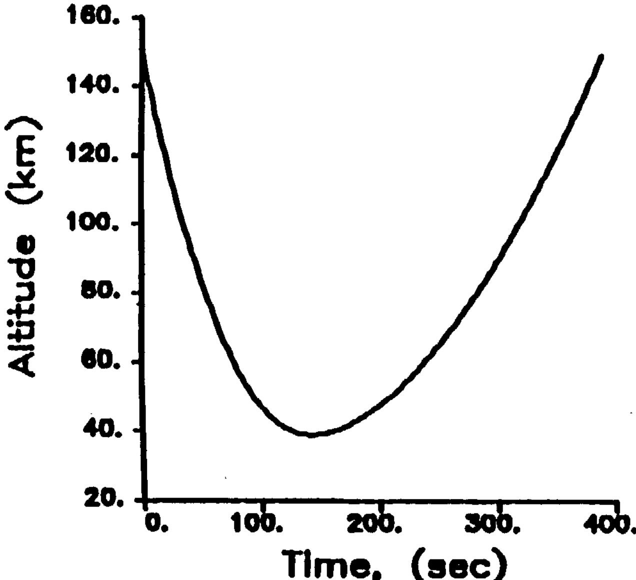
Mars Aerocapture Stagnation Point Heating

Typical aeroheating results are shown here. The high radiative heating spike exceeds the capabilities of present-day reradiative materials.

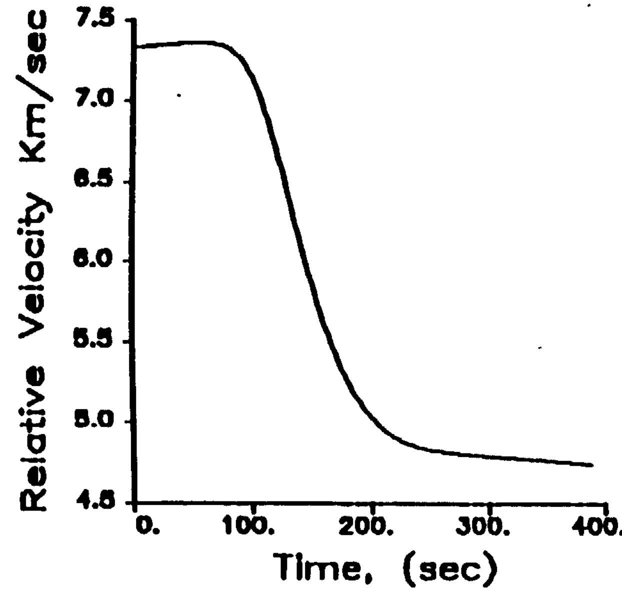
We have recently become aware of alternative radiative heating prediction methods that predict significantly less radiation under these conditions. The lesser predictions are within the capabilities of present-day materials. We will provide a comparison of these predictions in our April 1990 monthly progress report.

Mars Aerocapture Stagnation Point Heating

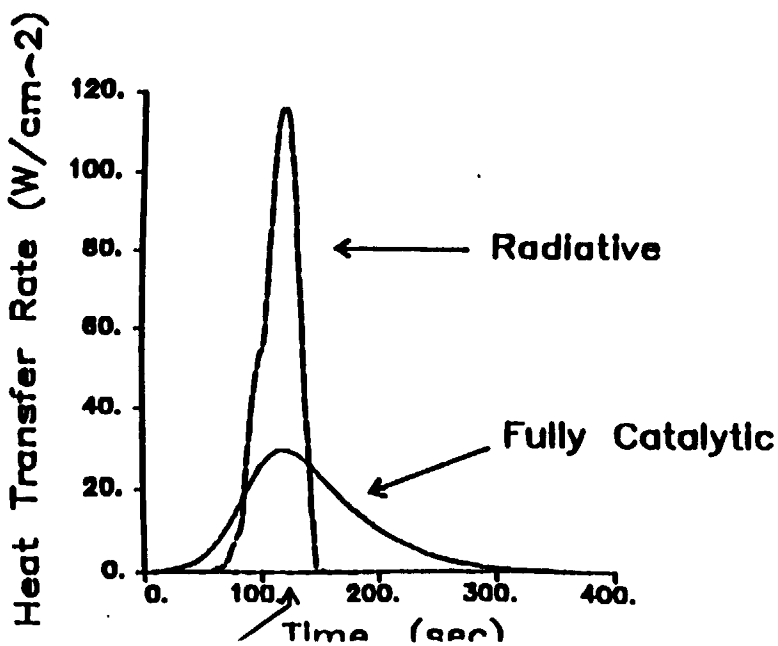
BOEING



Hyperboloid 0.5
L/D Aerobrake



C3 Of 30 km~2/sec~2
Bal. Coef. Of 39 g/cm~2
Averaged L/D of 0.0



Maximum Total Heat Flux
at 120 sec

Heating Distribution

This chart shows predicted heating distributions, displayed as adiabatic wall temperature, using the techniques described earlier. Values are in degrees K; under these predictions, most of the brake surface exceeds the capabilities of current materials, which is approximately 1900K.

Heating Distribution

Trajectory

- C3 = 30 km^2/sec^2
- Flight Averaged Lift = 0
- L/D = 0.5
- Ballistic Coefficient = 39.4 g/cm^2
- Max G-level = 3.0

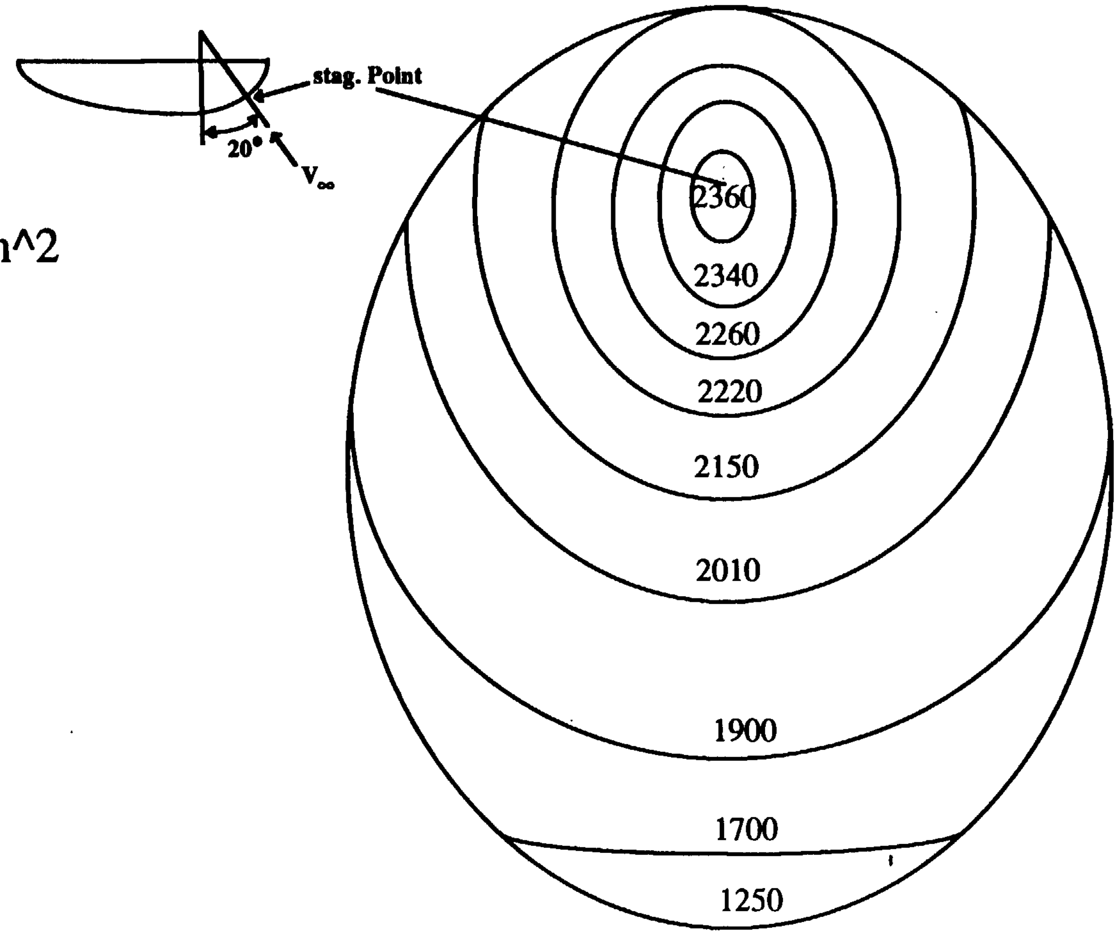
Conditions at Max Heat Rate

- Time = 120 sec
- Velocity = 6.714 km/sec
- Altitude = 41.2 km
- Density = 4.78×10^{-7} g/cm^2
- Total Heat Rate = 146 w/cm^2
- Stagnation Temp. = 2383 K

Total Heat Load

- Q = 7,601 j/cm^2

Emissivity = 0.8



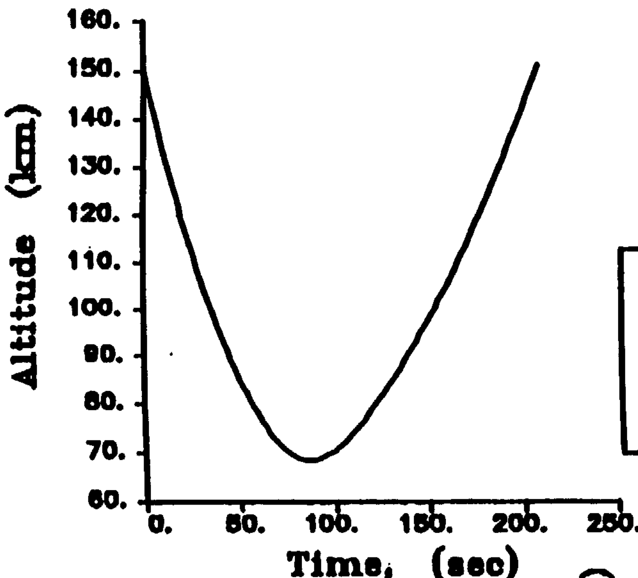
Temperature in K

Mars Return Aerocapture Stagnation Point Heating

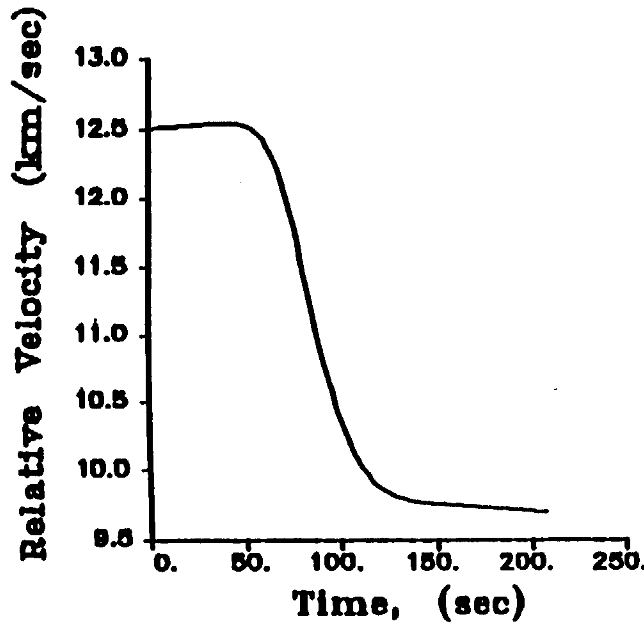
Results for Earth entry of the Earth Crew Capture Vehicle are shown here. The Earth Return C3 was approximately 50. Since the entry trajectory used an up-directed lift coefficient of 0.2, this is a relatively high-q, high-heating case.

Mars Return Aerocapture Stagnation Point Heating

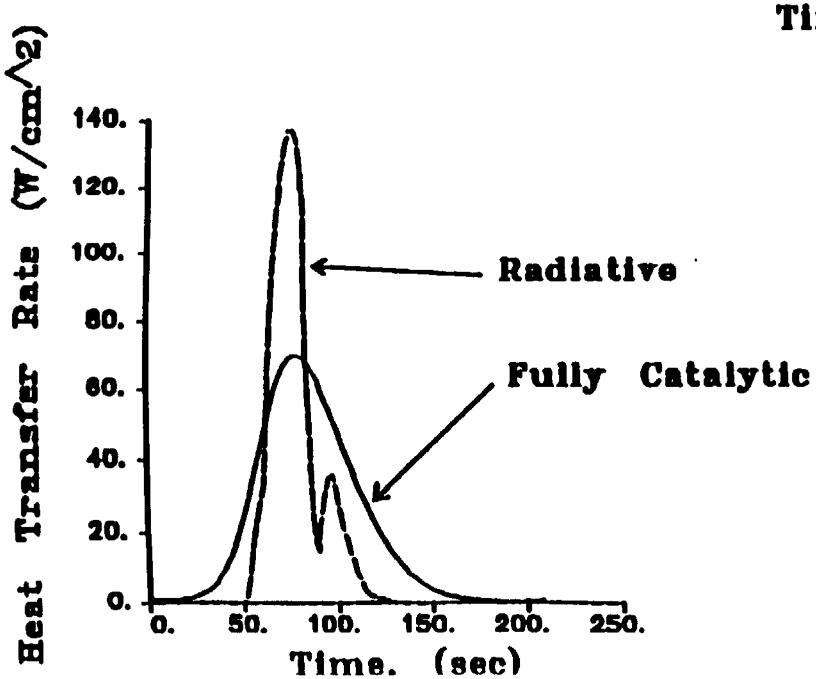
BOEING



Entry Velocity
of 12.98 km/sec
L/D of +0.2
Ballistic Coef.
of 9.765 g/cm^2



Data for the Crew Capture Vehicle



Aeroheating Principal Findings

The findings from our current analysis are that Mars capture aeroheating is a significant problem, unless correlations that predict less radiative heating can be verified. Several work-arounds are noted on the chart. In the next three months, we will be working with alternative radiation correlations and exploring the efficacy of these work-arounds.

Aeroheating Principal Findings

- **Mars Aerocapture Radiation Problems**

  - For C3's from 30-50 km^2/sec^2, Radiative flux is 80-90% of total heat flux.

- **Stagnation Temperatures for C3's ≥ 30 are above near term reradiative technologies.**

eg.

C3	Q(w/cm^2)	T °K
30	146.	2383
40	299.	2850
50	481.	3210

Note: 1993 reradiative technology ≈ 68 w/cm^2 or ≈ 1968 ° K

- **Options**

  - Use of ablators for Mars Aerocapture.
  - Improve reradiative materials.
  - Limit the Missions to lower approach C3's.
  - Modify or Change the Aerobrake shapes.
  - Optimize Trajectories for miminal aeroheating (Down-Lift).

- **Needs**

  - Improved Analysis for Non-Equillibrium Radiation in CO_2 Atmosphere
  - Engineering Methods for treating Non-Axisymmetric blunt body flows.

Importance of Landing Site Analysis

The reasons for performing landing site analyses are indicated on the facing page. Landing site access will be the requirements driver for Mars Excursion Vehicle aerobrake L/D and descent profile design.

Importance of Landing Site Analysis

- Landing site location determines the L/D requirements to get from orbit to the site
- L/D requirements determine the configuration of the Mars Excursion Vehicle Aerobrake
- The configuration of the aerobrake determines the load points, vehicle stress and available wake cone area to place the lander vehicle inside of
- The wake cone , stress points,and load points determine the configuration of the lander
- The size and shape of the aerobrake can determine the amount of packaging required for ETO launch and the number of launches required if it is assembled in space

Preliminary Mars Landing Sites Between ± 20° Latitude

The next three pages show a sampling of landing sites of scientific interest in the ± 20° latitude band on Mars. Altitudes are also shown, since altitude has a strong effect on landing delta V. We are presently designing for access to any site within this latitude range, at altitudes up to 5 km. with the COSPAR low-desnity atmosphere. This will permit landings up to 8 - 9 km altitude with typical atmosphere densities.

Preliminary Mars Landing Sites Between +/- 20° Latitude

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
Tharsis Montes	5°N	100°	9 km	Ascraeus and Pavons Mons, rill formations, Tharsis Thoius, unnamed crater
	10°N	82°	3-2 km	Tharsis Thoius, Echus Chasma, Fesenkov Crater, head of Kasei Vallis, Lunar Planum (colored soil)
	-10° S	137°	4 km	Mangala Valles, Memnonia and Sirenum Fossae, edge of Tharsis Montes shield, Aganippe Fossa, Arsia Mons Colored sands
	-15° S	115°	8-9 km	Arsia Mons, Noctis Labirinthus, Syria Planum, Claritas Fossae, crater area
Siniai Planum	-18°S	76°	1-8 km	Melas Chasma of Valles Marineris (possible access to Valles Marineris floor); Felis, Melas and Solis Dorsii, crater (unnamed) with rills/flows, Lassell Crater, Coptates Catena

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Preliminary Mars Landing Sites Between +/- 20° Latitude page 2

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
South of Eos Chasma	-19°S	49°	3 km	Eos Chasma (part of Valles Marineris, with possible access to the valley floor, accessible places in the valley floor -1 and -2 km) Lassell and Richey Craters, Felis Dorsa, crater field some with flow fields, Holden Crater
Hesperia Planum	-16°S	253°	4 km	Tyrrhena Patera (massive flow field from a single source), crater fields, surface cracks and fissures, Terra Tyrrhena area, small mounts
Elysium-Amazonis	0°	180°	0 km	Pettit Crater, Nicholson Crater, surface cracks, Orcus Patera, Cerberus Rupes, colored soils, old craters, Apollonaris Patera, Gusey Crater and flow field, edge of Elysium flow shield, Medusae Fossae, "new" craters in the Elysium flow shield

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Preliminary Mars Landing Sites Between +/- 20° Latitude page 3

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
Elysium Planitia	19° N	197.5°	2-3 km	Elysium Mons, Elysium Fossae, Ocrus Patera, Cerberus Rupes, Lockyer Crater, Phlegra Mons, colored sands, craters, old and "new"
Amazonis Planitia	15°N	155°	0-3 km	flow area around Olympus Mons, edge of Gordii Dorsum and Eumenides Dorsum formations, crater area east of Pettit Crater
Chryse Planitia	18° N	45°	0-(-1) km	Cryse depression (-3 km), end of Kasei Vallis, Sharonov Crater,Lunae Planum; Nanedi, Shalbatana , Simud, and Tibu Valles, end of Ares Vallis, craters, colored sands

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Preliminary Mars Landing Sites Between +/- 20° Latitude page 4

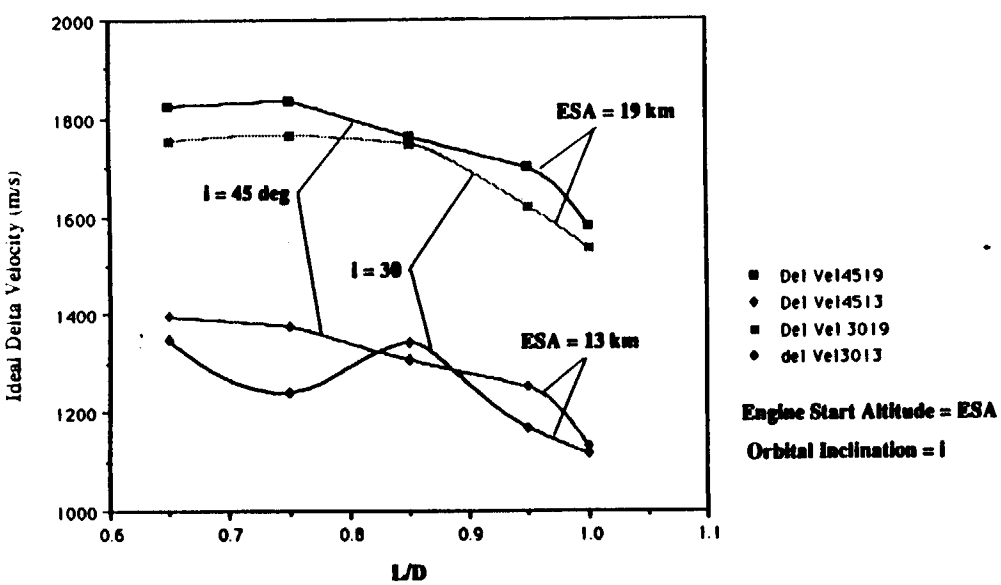
Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
North of Ganges Catena	-2°S	68°	1 km	Ophir Chasma (part of Valles Marineris, with possible access to the valley floor), Hebes Chasma, Echus Chasma, Juveniae Chasma, Ophir Planum, Lunae Planum, crater field, colored soil
Elysium Planitia	19°N	226°	0 km	Hephaestus Fossae, Elysium Fossae,Elysium Mons, Albor Tholus, Eddie Crater with interior formation, colored sands

Landing Cross-Range Analysis

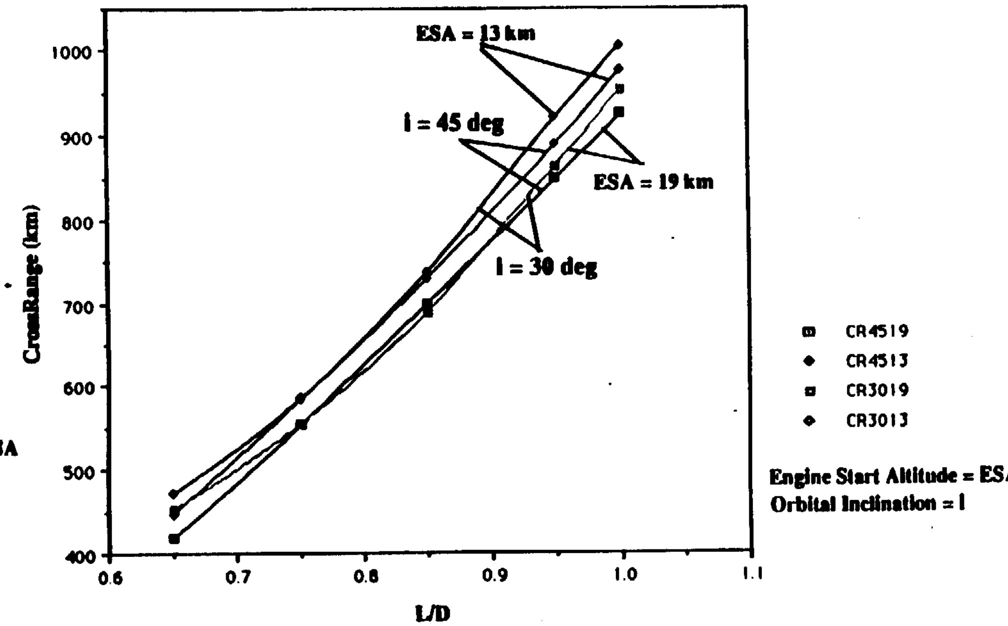
Shown here are results of a parametric cross-range versus L/D analysis. These simulations did not use the aerodynamic flare; the effect on cross-range is nil but the landing delta V is reduced by about 400 m/sec.

Landing Crossrange Analysis (5km Landing Altitude, Low Density Atmosphere)

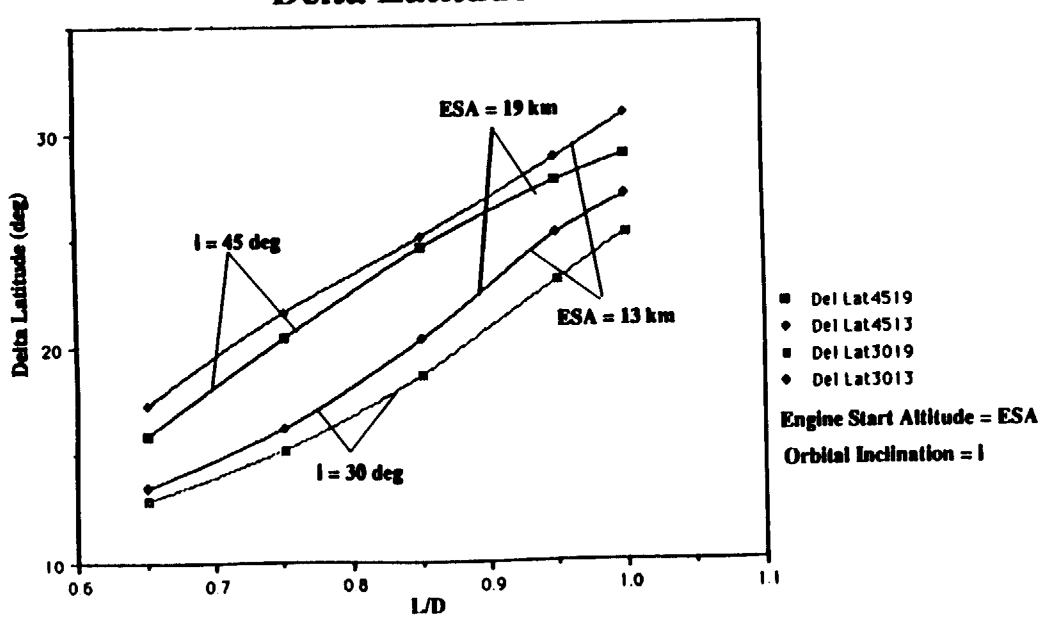
Ideal Delta Velocity



Crossrange



Delta Latitude



Mars Descent Analysis Findings

Results of the Mars descent analysis are summarized here.

Mars Descent Analysis Findings

- Wide variation in vehicle crossrange achievable with $0.5 < L/D < 1.0$

- Latitudinal displacement of > 20 deg is achievable for a wide range of entry conditions, for $L/D > 0.9$

- Increasing descent vehicle L/D provides diminishing Delta V requirements

- Landing altitudes of 5 km to 10 km above Mars reference are achievable with present crossrange requirements, for $L/D > 0.95$

- High L/D shape will be a flatter shape then previously investigated with a partial top shield to control wake impingement

- A flare aeromaneuver will be performed at the end of the landing sequence (before vehicle landing for the reusable aerobrake and aerobrake drop for the non-reusable Level II scenario). This flare will be controlled by:
  - For the non-reusable aerobrake, a large flap(.25 of aeroshell area) will separated prior to aerobrake drop.
  - For the reusable aerobrake, the center of gravity control, and therefore the flare maneuver, will be managed by pumping LOX to and from main and auxiliary tanks
  - The reusable areobrake will also have an articulated flap with a maximum area of 10% of the aerobrake

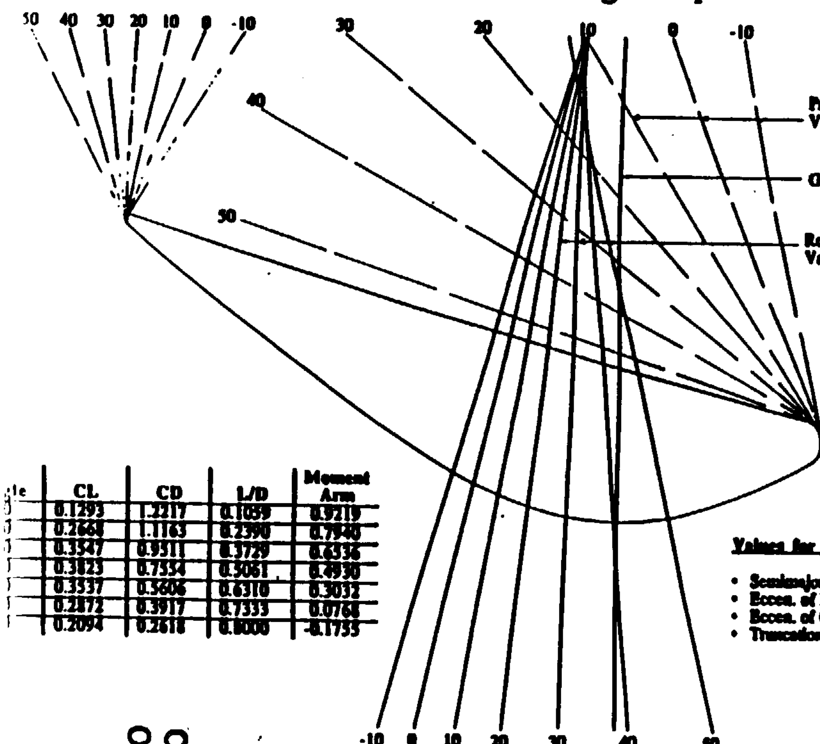
Aerobrake Shapes

A high L/D shape is compared here to the L/D 0.5 shape. The need for pitch control leads to a further flattening of the shape; even the high L/D shape shown here requires a very large flap to obtain the needed pitch control.

Aerobrake Shapes

L/D 0.5

Aerobrake Lift and Drag Properties



...le	CL	CD	L/D	Moment Arm
[illegible]	0.1293	1.2217	0.1059	0.9219
[illegible]	0.2668	1.1163	0.2390	0.7940
[illegible]	0.3547	0.9511	0.3729	0.6336
[illegible]	0.3823	0.7554	0.5061	0.4930
[illegible]	0.3537	0.5606	0.6310	0.3032
[illegible]	0.2872	0.3917	0.7333	0.0768
[illegible]	0.2094	0.2618	0.8000	-0.1755

Values for Shape Parameters

- Semimajor Axis Ratio 2.00
- Eccen. of Body Revol. 1.60
- Eccen. of Cutting Cyl. 0.40
- Truncation/SMA Ratio 0.00

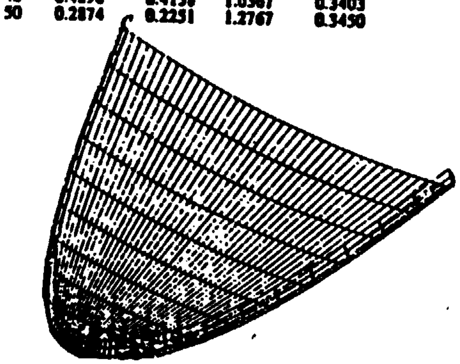
L/D 1.0

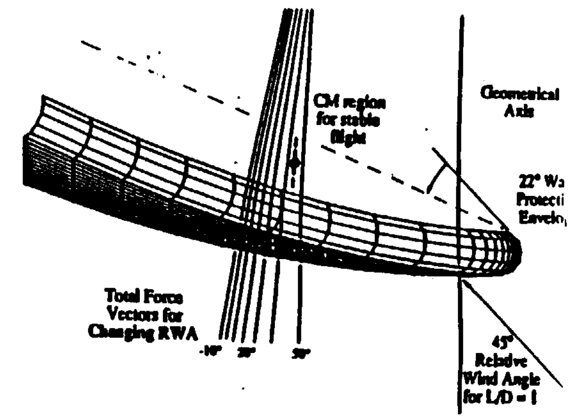
High L/D Open Shape

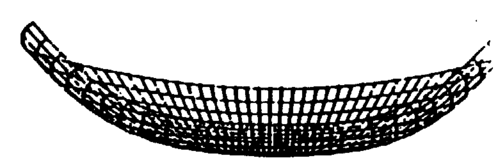
Values for Shape Parameters

Semimajor Axis Ratio	2.0000
Eccen. of Body of Revol.	1.8000
Eccen. of Cutting Cyl.	1.0500
Truncation/SMA Ratio (lip 0.025 SMA)	0.5000
Plan Area	1.2684
Surface Area	1.4753

Angle	CL	CD	L/D	Moment Arm
-10	0.0353	1.4771	0.0239	0.3726
0	0.2680	1.3962	0.1920	0.3652
10	0.4445	1.2063	0.3685	0.3581
20	0.5326	0.9463	0.5628	0.3510
30	0.5225	0.6659	0.7847	0.3445
40	0.4290	0.4138	1.0367	0.3403
50	0.2874	0.2251	1.2767	0.3450







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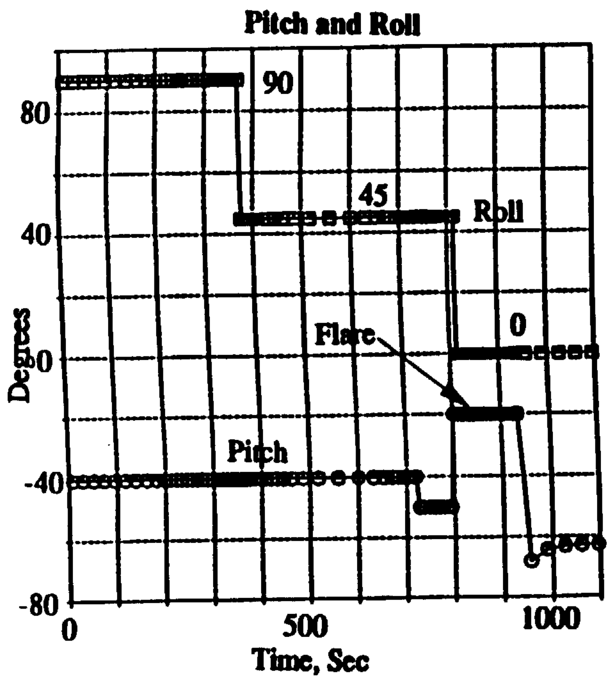
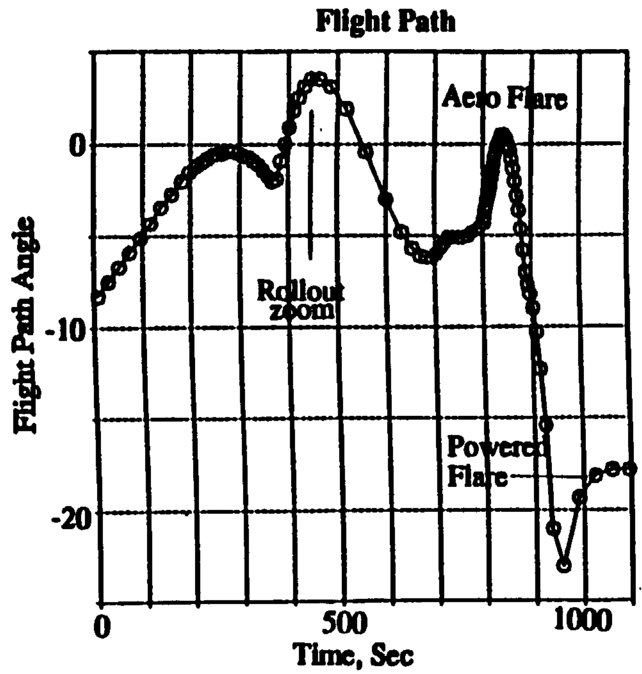
Mars Landing Simulation with Aerodynamic Flare

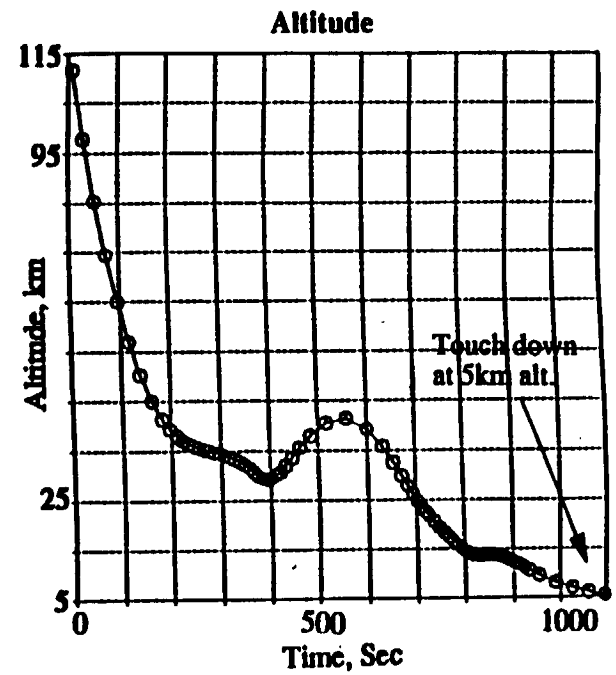
Shown on this chart are results of an un-optimized, but typical, aerodynamic and propulsive descent. Mars entry occurs at a 90° roll angle to obtain maximum cross-range. As the vehicle slows to below circular velocity, roll-out in two steps maintains roughly level flight. Most of the descent is flown at L/D = 1. Prior to engine start, the L/D is briefly increased (drag decreased) to increase speed. Then the vehicle is pitched to maximum lift coefficient at L/D about 0.5. This causes an aerodynamic flare, decreasing speed and increasing path angle. The result is a significant decrease in rocket thrust and delta V for landing.

The importance of this is that it generates a requirement for pitch control, a requirement not present for aerocapture. The combination of high L/D and pitch control will lead to selection of an aerobrake shape much different from the MTV aerocapture case.

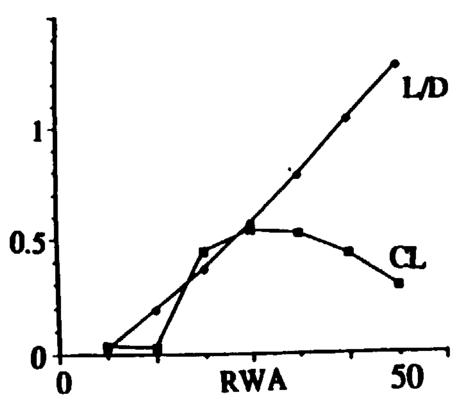
Mars Landing Simulation With Aerodynamic Flare







- COSPAR low-density atmosphere
- Entry mass 81t.
- Thrust 80k
- Max L/D 1.1
- Ref Area 750m^2



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Radiation Protection

Matthew Appleby

ADVANCED
CIVIL
SPACE
SYSTEMS

Agenda

- Requirements
- Crew Exposure Guidelines
- Environments and Mission Phase Relationships
- Protection Concepts
- Research Concerns
- IR&D Objectives

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Radiation Protection Requirements Summary

- Provide protection from natural and manmade radiation environments (ionizing and nonionizing) to crew members and spacecraft electronics
- Protection to be provided as stipulated in NASA STD-3000 section 5.7.2.2.2, paragraph *a*, with the addition of real-time monitoring *
- Radiation protection shall not be provided in the MEV *
- Radiation protection for crew members to conform to method of ALARA
- "Guidance on Radiation Received in Space Activity": NCRP Report No. 98 dated July 31, 1989 (Dosage levels permitted) shall be adhered to *

* From Level II - Human Exploration Study Requirements: September 8, 1989

Units and Terms Used to Describe Human Responce to Ionizing Radiation

The quality factors, Q, have been set by the International Commission on Radiological Protection (ICRP) and accounts for the different biological effectiveness of various ionizing radiation. The values that have been assigned to the high atomic number and energy particles (HZE) associated with galactic cosmic radiation and solar proton events reflects a conservatism. This conservatism is dictated by a serious lack of knowledge about the biological effectiveness of the high LET radiations (radiations with LET's ≥ 175 keV/μm). Average Q values for these particles may be exaggerated. Recent research has raised questions as to whether such high quality factors are justified. For example, the dose equivalence (DE) which appears on the following chart, assumes a uniform distribution of energy throughout the tissue of interest. In reality less than half of the cells of an astronaut will be traversed by HZE particles. A more realistic approach may be to assign relative health risks per fluence (particles /cm^2•s) of given linear energy (or charge and velocity).

Units & Terms Used to Describe Human Response to Ionizing Radiation

Dose - (D)

- The amount of radiation energy absorbed by tissue
- Common unit of measure - rad (1 rad = 100 ergs per gram of material)
- SI unit for dose - gray (Gy)
- 1 Gy = 100 rads

Linear Energy Transfer - (LET)

- Denotes the rate of energy dissipation along the path of a charged particle
- Units expressed in energy/unit length (keV/µm)

Quality Factor - (Q)

- An artificial factor dependent on the LET of which biological effects from absorbed doses may be related to X- and gamma radiation (how much biological damage)
- Nondimensional factor
- Values are based on the most detrimental biological effects from continuous low dose exposure
- Values for many high rate exposures may be considerably lower

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Units & Terms Used to Describe Human Response to Ionizing Radiation

(continued)

Dose Equivalent - (DE)

- The amount of biologically damaging ionizing radiation
- Common unit of measure - rem (roentgen equivalent man)
- SI unit - sieviert (Sv)
- 1 Sv = 100 rem
- $DE = D \cdot Q$

Relative Biological Effectiveness - (RBE)

- Related but distinctly different form Q
- Based solely on experimentally determined effects of different types of radiation on biological systems
- Nondimensional quantity

Short Term Dose Equivalent Limits and Career Limits for Protection Against Nonstochastic Effects

This chart provides the latest recommended dose equivalent limits for astronauts contained in the NCRP Report No. 98, Guidance on Radiation Received in Space Activities. The NCRP (National Commission on Radiation Protection) recognizes the many inherent risks involved in exploratory class missions that leave the protective confines of the Earth's magnetosphere. "No specific limits are recommended for personnel involved in exploratory class missions, for example to Mars". The NCRP recommends in addition to the principal of ALARA (As Low As Reasonably Achievable), the career limits proposed be adhered to as guidelines rather than limits whenever possible. NASA has a radiation protection program for astronauts that limits the amount of radiation received deep in the body to what is judged an acceptable level. Ancillary standards to the eye and skin are also shown. In certain situations such as EVAs in the trapped radiation belts, the dose to the eyes or skin could be very high before the dose limits to the BFO (Blood Forming Organs) could be met. Thirty day limits are set to avoid immediate radiological impacts on a mission involving nausea, vomiting and the like. The career dose-equivalent limits are based upon keeping the life-time risk of excess cancer mortality to less than 3%, an excess risk judged to be acceptable. As can be seen the career limits differ according to sex and age.

Short Term Dose Equivalent Limits and Career Limits for Protection Against Nonstochastic Effects

All values presented in Sv - (1 Sv = 100 rem)

Time Period	BFO*	Lens of Eye	Skin
30 day	0.25	1.0	1.5
Annual	0.5	2.0	3.0
Career	See table below	4.0	6.0

* Blood forming organs. This term has been used to denote the dose at a depth of 5cm

Career whole body dose equivalent limits based on a lifetime excess risk of cancer mortality of 3%

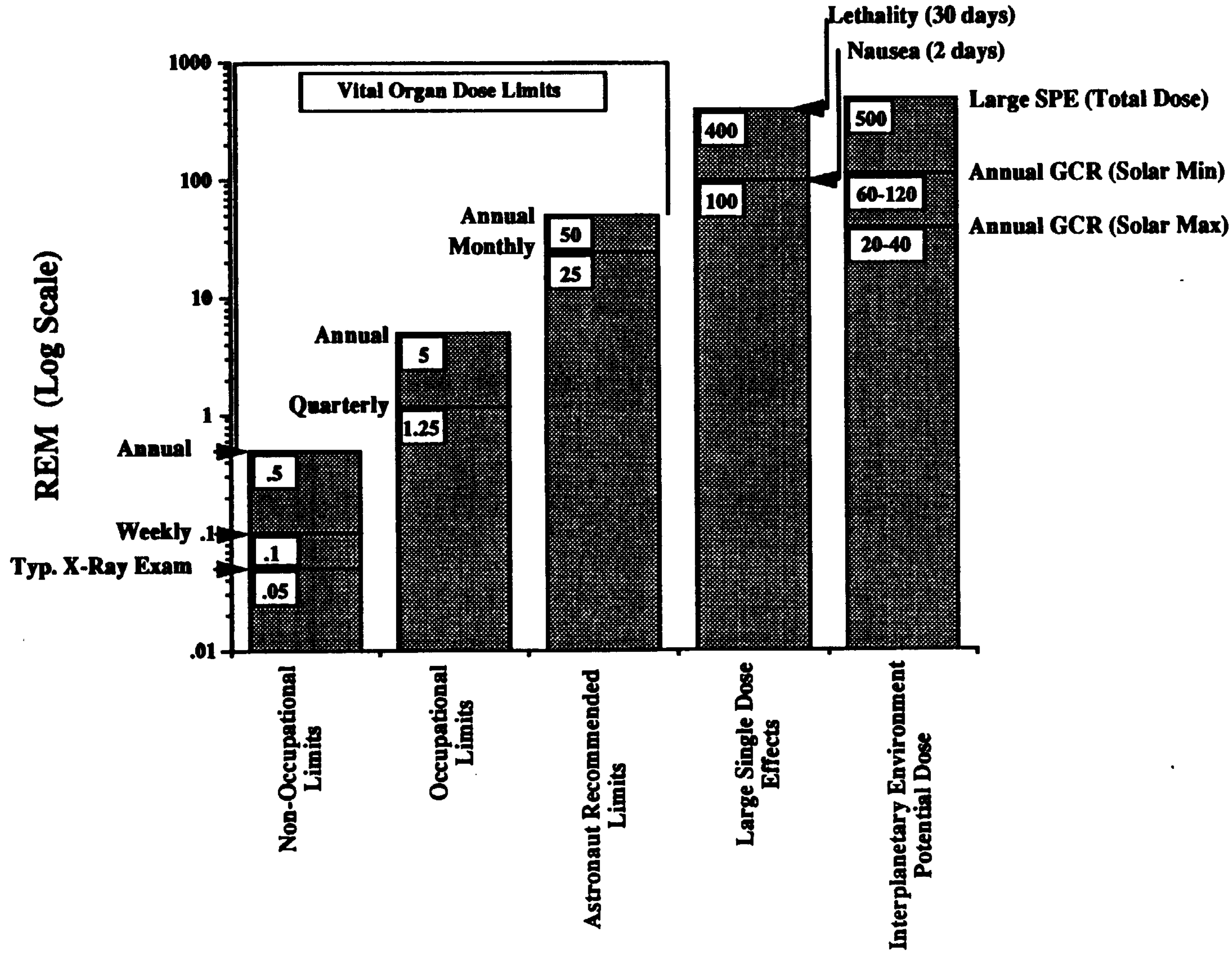
Age (years)	Female	Male
25	1.0	1.5
35	1.75	2.5
45	2.0	3.2
55	3.0	4.0

• Data from Guidance on Radiation Received in Space Activities, NCRP Report No. 98

Radiation Dose Comparison - Cause/Effect/Limits

This chart presents a comparison of acceptable equivalent dose limits for terrestrial non-occupational and occupational workers and astronauts to the blood forming organs. Exposure of crew members in space may be reduced but cannot be completely eliminated and, therefore, must be considered an occupational hazard. However, for various reasons, occupational standards that are used on the ground should not be applied directly to situations in space. In the recommendation of the career exposure limits by the NCRP, cancer is considered the principal risk. Based on this consideration the NCRP recommends a career limit of 3% excess risk of cancer mortality for space activities for both sexes of all ages. In addition to the comparison between occupations, large single dose effects are represented and the potential interplanetary environment dosages that may be encountered on a trip to Mars. The chart may be somewhat deceptive due to logarithmic scale that is used. This is a reconstruction of a chart presented by Dr. S. Nachtwey in "Health Physics", August 1988.

Radiation Dose Comparison - Cause/Effect/Limits



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Dose comparison data from Dr. S Nachtwey, report from "Health Physics, August 1988

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Radiation Dose Examples and Effects

From Life on Earth	Exposure
• Transcontinental round trip by jet	0.004 rem
• Chest x-ray (lung dose)	0.010 rem
• Living one year in Houston	0.100 rem
• Living one year in Denver	0.200 rem
• Xeromammography (breast dose)	0.383 rem
• Barium enema (intestine dose)	0.875 rem
• Living one year in Kerala India	1.300 rem
• Max. allowable radiation worker/yr	5.000 rem
Manned Spaceflight	
• Skylab 3, 84 days (blood forming organs)	7.94 rem
(eye lens)	12.83 rem
(skin)	17.85 rem
• Max. allowable space worker/yr	50.00 rem

Effect in Healthy Adults	Acute Dose
• Blood count changes common	50 rad
• Vomiting, "effective threshold"	100 rad
• Mortality, "effective threshold"	150 rad
• LD_{50} minimal medical treatment	320-360 rad
• LD_{50} supportive medical treatment	480-540 rad
• LD_{50} bone marrow/blood stem cell transplant	1000 rad
Effects on Reproductive Systems	
• 50% temporary sperm count reduction	15 rad
• 100% sperm loss lasting a few months	100 rad
• Male sterility lasting 3 or more years (if subject survived high dose)	600 rad
• Possible menopause in 40 yr.-old woman	300 rad
• Possible temporary menstrual suppression in 20 yr.-old woman.	300 rad

* From the SICSA Outreach Journal; Vol. 2,No. 3, July-September, 1989, Stuart Nachtwey

Quality Factor for Various Types of Radiation

These two charts show the relation between quality factor (Q) and linear energy transfer (LET). The Q values are those which are currently used for various types of radiation. As a given particle degrades in tissue, the quality factor will rise as its energy transfer per micron rises. For a beam of protons having a wide range of energies, the average Q tends to drop with increasing depth in tissue as the lower energy component tends to be removed with increasing depth and the high-energy component continues its traversal.

The standard Q values are based on the most detrimental chronic biological effects for continuous low-dose rate exposure that may be met in industrial situations.

Quality Factor for Various Types of Radiation

Type of Radiation	Quality factor, Q
X-rays	1
Gamma rays & bremßtrahlung	1
Beta particles, electrons, 1.0 MeV	1
Beta particles, 0.1 MeV	1
Neutrons, thermal energy	2.8
Neutrons, 0.0001 MeV	2.2
Neutrons, 0.005 MeV	2.4
Neutrons, 0.02 MeV	5
Neutrons, 0.5 MeV	10.2
Neutrons, 1.0 MeV	10.5
Neutrons, 10.0 MeV	6.4
Protons, greater than 100 MeV	1 - 2
Protons, 1.0 MeV	8.5
Protons, 0.1 MeV	10
Alpha particles (helium nuclei),5 MeV	15
Alpha particles, 1 MeV	20

LET - Q relationship	
LET - in water (keV/μm)	Q
≤ 3.5	1
7	2
23	5
53	10
≥ 175	20

Nature and Location of Electromagnetic and Particle Ionizing Radiation

Ionizing radiations vary greatly in energy. Electromagnetic radiation have energy quanta determined by their wavelength or frequency. The energy of particulate radiation depends on the mass and velocity of the particles. This chart summarizes the main types of ionizing radiation including their charge, mass, and location. Crew members will be subjected to radiation emanating from two primary sources, those that are manmade and those originating from natural sources. Naturally occurring radiation is comprised of charged particles and accompanying electromagnetic radiation attributable to a number of distinct sources.

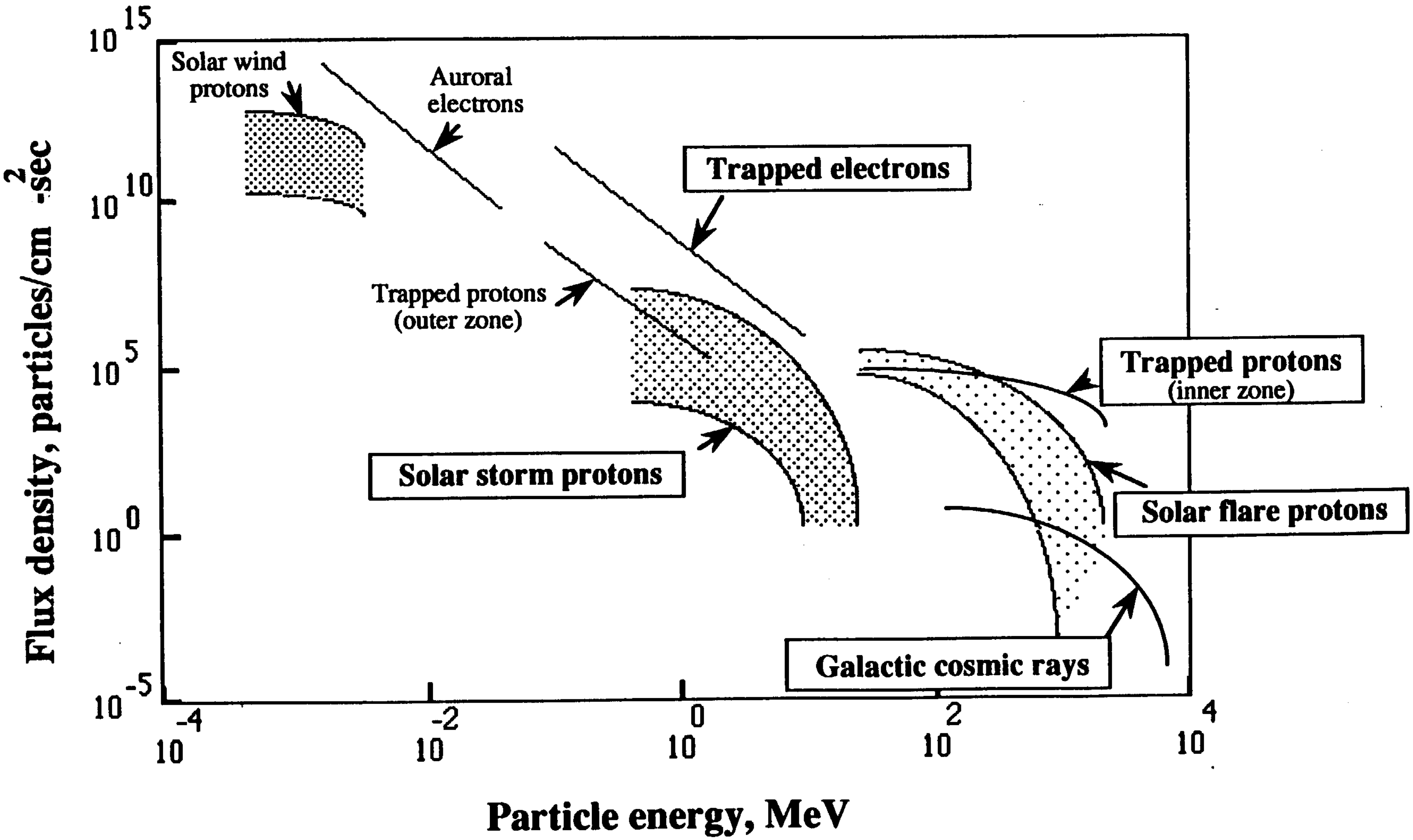
Nature and Location of Electromagnetic and Particulate Ionizing Radiation in Space

Name	Charge	Nature of radiation	Mass	Location/source
X-ray	0	Electromagnetic	0	Radiation belts, solar radiation (produced by nuclear reactions and by stopping electrons) Bremsstrahlung radiation (-e deflection by Coulomb field at atomic nuclei of target material)
Gamma ray	0	Electromagnetic	0	Everywhere in space (disintegration of atomic nuclei)
Electron	-e	Particle	1 m_e	Radiation belts and elsewhere
Proton	+e	Particle	1840 m_e or 1 am	Galactic and solar cosmic rays, radiation belts
Neutron	0	Particle	1841 m_e	Secondary particles produced by nuclear interactions involving primary particle flux
Alpha particle (helium nucleus)	+2e	Particle	4 am	Galactic and solar radiation
HZE particle (heavy primary)	≥+3e	Particle	≥ 6 am	Galactic and solar radiation

Space Radiation Environments

Particle radiations that occur in space are summarized in this chart. The various radiation environments occur with both temporal and spatial variations. Trapped particles exist only in the geomagnetosphere, the auroral electrons are observed only in the polar regions, and solar flare protons are emitted at dangerous levels infrequently and highly unpredictably. Radiations with energies below 100 keV and protons below approximately 10MeV are important primarily from a materials standpoint and are considered to be biologically unimportant. Galactic cosmic radiation, trapped protons and electrons within the radiation belts, and solar flare protons are all biologically very important. Even though the galactic cosmic radiation has a very low flux density many questions surround them because of their particular compoition and high energies.

Space Radiation Environments



From Workshop on the Radiation Environment of the SPS, J.W. Wilson

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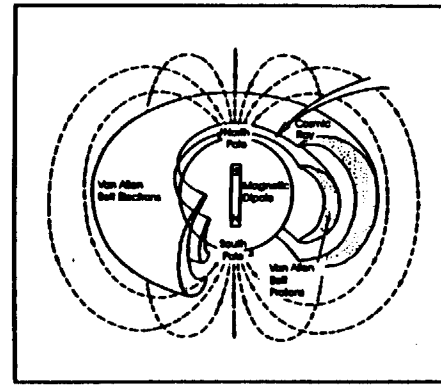
Radiation Environments



• Trapped Radiation Belts

~ Belts geomagnetically trapped around Earth

~ Inner proton belt
- Inner belt consists of protons and electrons
- Flux density and energy of protons dominate as source of ionizing radiation
- Inner belt densities respond to temporal variations in solar activity
- Extends out to an altitude of approximately 12000 km
- Proton density peaks at an altitude of 2000 km

~ Outer electron belt
- Consists primarily of trapped electrons
- Secondary radiation (Bremßtrahlung) dominates as source of ionizing radiation
- Outer belt also responds to temporal variations in solar activity
- Extends from an altitude of approximately 16000 to 36000 km
- Density peak on average at 20000 km

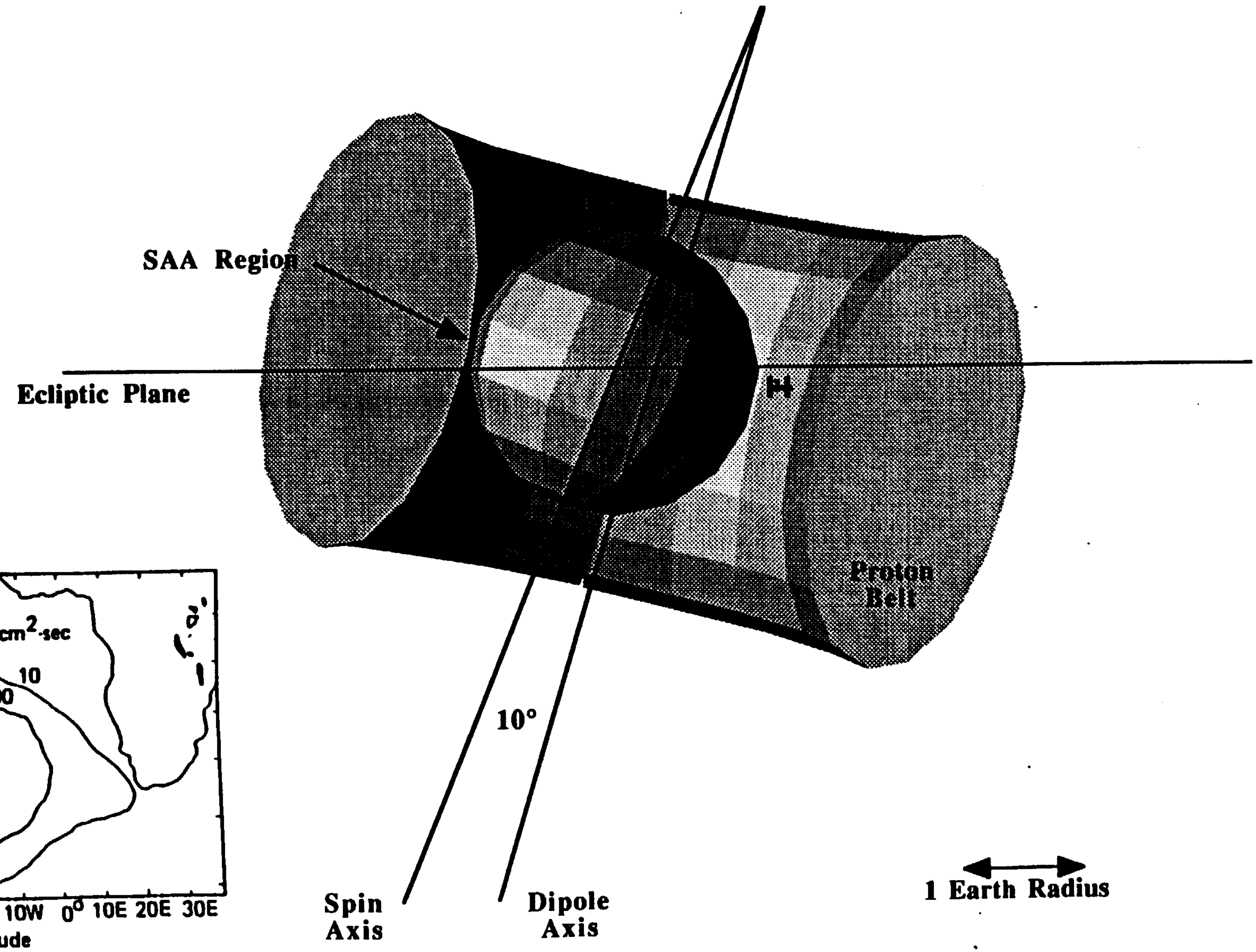
~ South Atlantic Anomaly
- Caused by combination of [1] anomaly in geomagnetic field over South Africa and [2] slight displacement of dipole axis (10°) from Earth's rotational axis
- Proton intensity for energies >30MeV are observed at altitudes between 200 and 400 km, approximately 1100 to 1300 km below normal

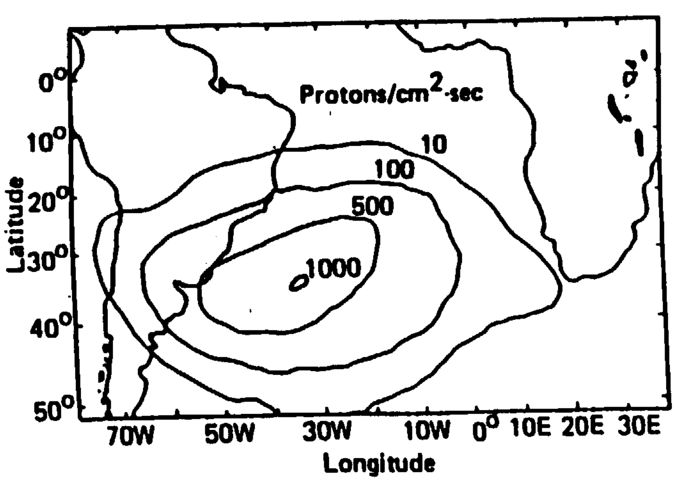
South Atlantic Anomaly

A feature of spatial distribution which has and is attracting much interest is the South Atlantic Anomaly (SAA). Caused by a combination of an anomaly in the geomagnetic field over South Africa and a slight displacement of the dipole axis (the magnetic central axis) from the Earth's rotational axis, the fluxes of the trapped particles are larger at low altitudes over the South Atlantic Ocean. The SAA extends from from 20° East to about 70° West longitude and 10° to 55° South latitude. The contours shown are trapped proton intensities for energies ≥ 30 MeV at an altitude of 200 km.

For trajectories of space vehicles of ~30° inclination, there will be five or six traverses through this region each day. Experience with Earth orbital missions to date indicates that nearly all of the accumulative radiation exposure has been attributable to passage through this zone. During the period of vehicle assembly and checkout this will be a concern to both crew and electronics.

South Atlantic Anomaly





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Radiation Environments (continued)

- **Galactic Cosmic Radiation (GCR)**

  ~ Originates outside of the solar system

  ~ Radiation consists of atomic nuclei ionized and accelerated to very high energies

  ~ Present isotropically in space

  ~ Decrease in flux caused by increase in strength of interplanetary magnetic field (below 100 Mev/nucleon)
  - flux density at solar: maximum = 2 prot/cm^2 ; minimum = 4 prot/cm^2

  ~ In the energy range from 100 MeV/nucleon to 10 GeV/nucleon, where fluence is greatest the baryonic component consists of:
  - 87% protons (H^+)
  - 12% alpha particles (H^{++})
  - $\leq$ 1% HZE particles (high Z, high energy)

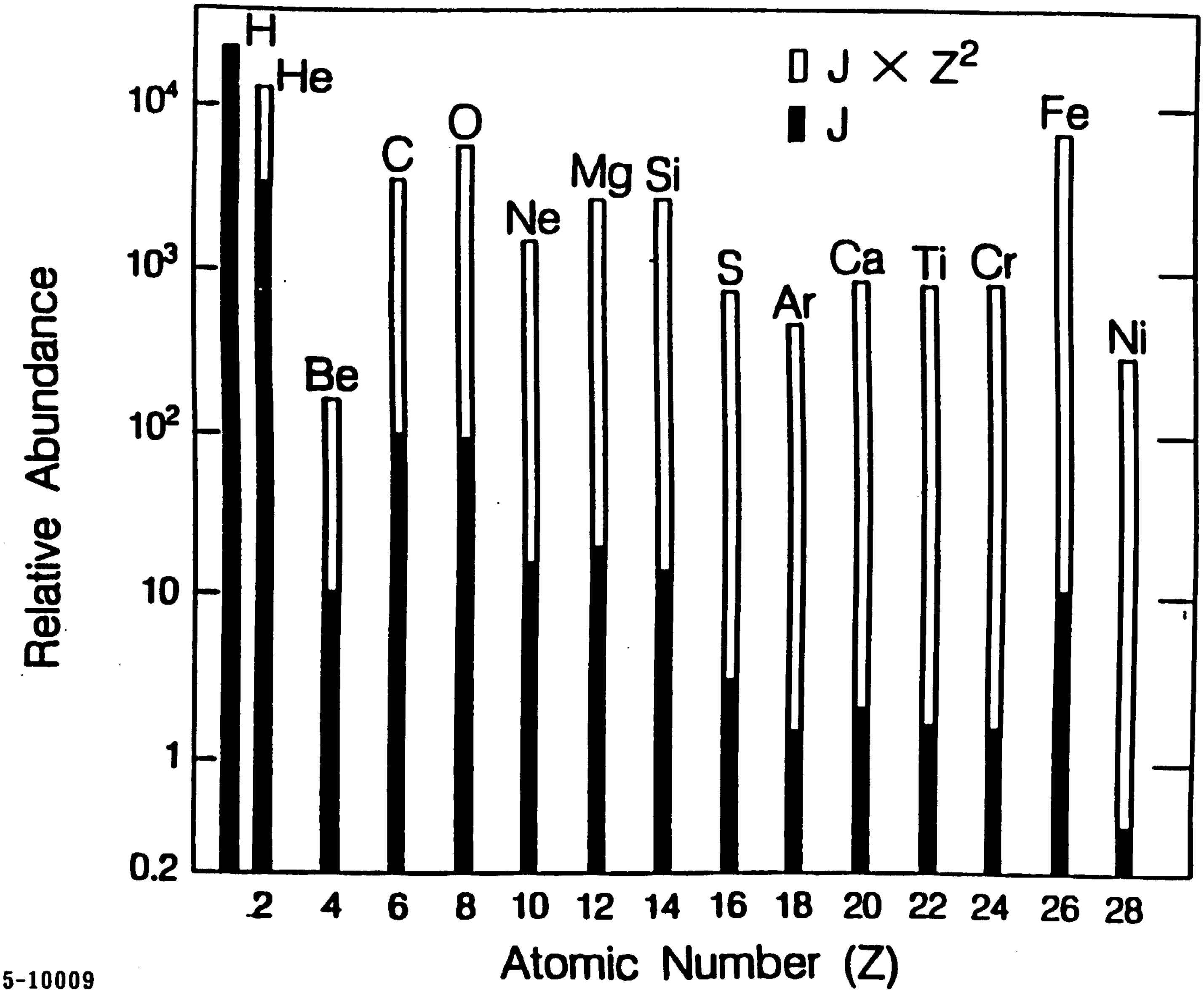
  ~ Main contribution to the radiation dose equivalent comes from the HZE particles and not from protons

  ~ Energies of particles extend to values of 10^{20} eV/nucleon

Relative Abundances of GCR Nuclei and a Measure of Their "Ionizing Power"

This histogram shows the relative abundances of the even numbered GCR nuclei (solid bars, J) compared to their abundances weighted by the square of the particles charge (Z^2) to give a measure of the "ionizing power" of each element (open bars). The ions that are heavier than helium are generically termed HZE particles. Although iron ions are only one-tenth as abundant as carbon or oxygen ions, their contribution to the GCR dose is substantial as indicated.

Relative Abundances of GCR Nuclei and a Measure of Their "Ionizing Power"



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From Guidance on Radiation Received in Space Activities; NCRP Report No. 98

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Radiation Environments (continued)

- **Solar Proton Events (SPE)**

  ~ Highly unpredictable in nature (frequency, intensity, duration)

  ~ Large emissions of charged particles [primarily: protons (95-98%), alpha (1-3%) and HZE (<1%)]

  ~ Large fluences of charged particles emitted from the sun primarily associated with solar flare activity

  ~ Occurrences of flares is associated directly with the 11 year solar cycle

  ~ Flares tend to occur more frequently during the declining portion of the 11 year cycle

  ~ Solar proton events fall into two broad categories
    - "ordinary" events
    - anomalously large events (ALSPE); on average may occur 2 or 3 times during 4 to 6 year period of high sun spot activity

  ~ Large solar flares can have fluences greater > 10^{10} protons/cm^2 with energies > 10 MeV

  ~ Potential of delivering extremely high dosages in short period of time

  ~ Small percentage of flares will be of sufficient intensity to emit large proton fluences

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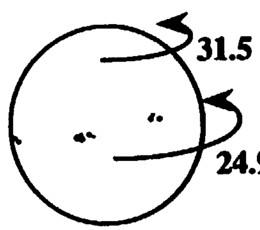
The Active Sun



Solar Differential Rotation

- Solar intensity will fluctuate rapidly due primarily to distortion of the Sun's large scale magnetic field

- Distortion of the magnetic field comes from differential rotation of gaseous body

- Magnetic field becomes twisted and contracted into specific regions such as facula, plage, spicules, prominences, sunspots, and flares

- Energy is often times released explosively in the form of a solar flare appearing as sudden local brightening in the chromosphere

- Stored magnetic energy is released as kinetic energy as field relaxes back to initial state (total energy released may be 10^{21} to 10^{25} joules integrated over three flare phases)
    - ~ precursor - slight enhancement of observed soft x-rays
    - ~ flash - increase in optical and x-ray emission by 50% above background
    - ~ main phase - bulk of energetic particle emission

- Radiation from solar flare extends from radio to x-ray wavelengths

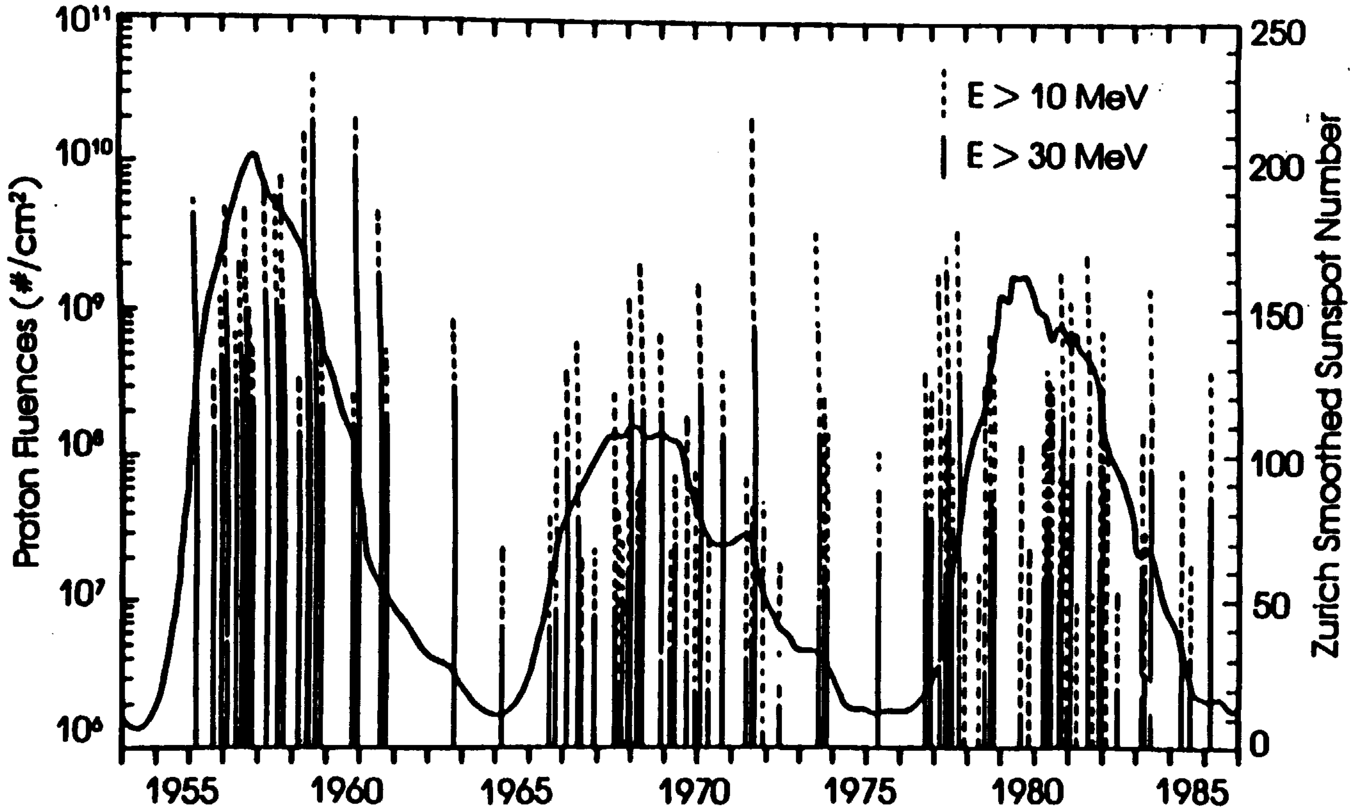
- Most flare events last about an hour. ALSPE, highly lethal occurrences are relatively rare but will last for hours or even days

Solar Activity and Flare Proton Fluence

It is important to note the effect of the solar cycle, or sunspot cycle, on the occurrence of solar-proton events. The sunspot number has been observed for approximately 200 years and varies with an average period of 11 years. During the upper half of the cycle, when the sunspot number is the largest, solar-proton events occur. They generally do not occur during the lower half of the cycle. Attempts to obtain detailed relationships between sunspot numbers and the frequency have shown that nothing can be said with the assurance beyond the fact that events tend to occur during the upper half of the cycle.

This chart provides a basis of comparison between the Zurich Smoothed Sunspot Number, the proton fluences and the time of occurrence for solar cycles 19, 20, and 21 from left to right. Currently cycle 19 is considered one of the most extreme cases in terms of sunspot number. The occurrence of several SPE's during this cycle (i.e., 2/56 and 11/60) were used as the basis for modeling protective measures for early manned missions. These events were used in fact as "worst case flares" until the occurrence of the 1972 event in August. Occurring during a cycle that was initially thought to be "average", it became apparent as to the lack of understanding we had in making predictions of such events. We are aware today that had the 8/72 event occurred at a more "favorable" location on the solar surface relative to the Earth this event would have been substantially larger.

Solar Activity and Flare Proton Fluence



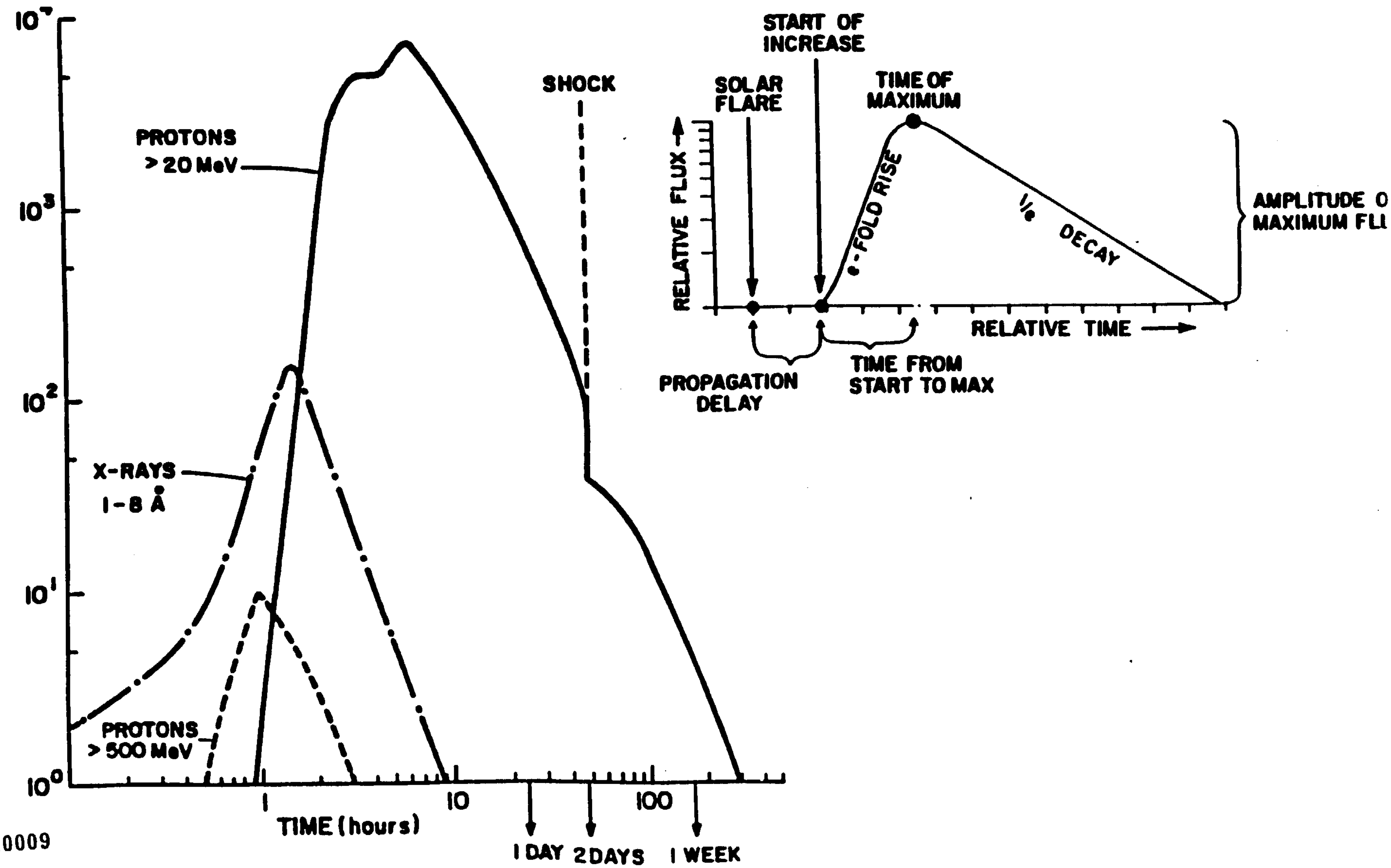
* From the SICSA Outreach Journal; Vol. 2,No. 3, July-September, 1989, Stuart Nachtwey

Relative Time of Solar Particle Emissions at 1 AU

It is important for the purpose of evaluating potential radiation protection schemes to understand solar particle propagation. Energetic solar particles will reach the orbit of Earth in a few short minutes if the particles have high energies, or within hours if possessing lower energies. This chart presents a relative time scale of solar emissions at 1 AU.

The inset graph shows the general time behavioral characteristics of a solar proton event. The propagation delay time is defined as the time from the maximum of the visible flare intensity to the particle arrival at the detector. The delay time will vary considerably from event to event with variations from several minutes to hours. The fold rise is the time interval between the first arrival of the particles of a particular energy and the time at which the flux of these particles reaches its maximum intensity. The fold rise is also strongly event and energy dependent, the high-energy having a shorter rise time, again times vary from minutes to hours. Finally the decay time is that time between maximum flux intensity and the disappearance of particles of a given energy.

Relative Time of Solar Particle Emissions at 1 AU



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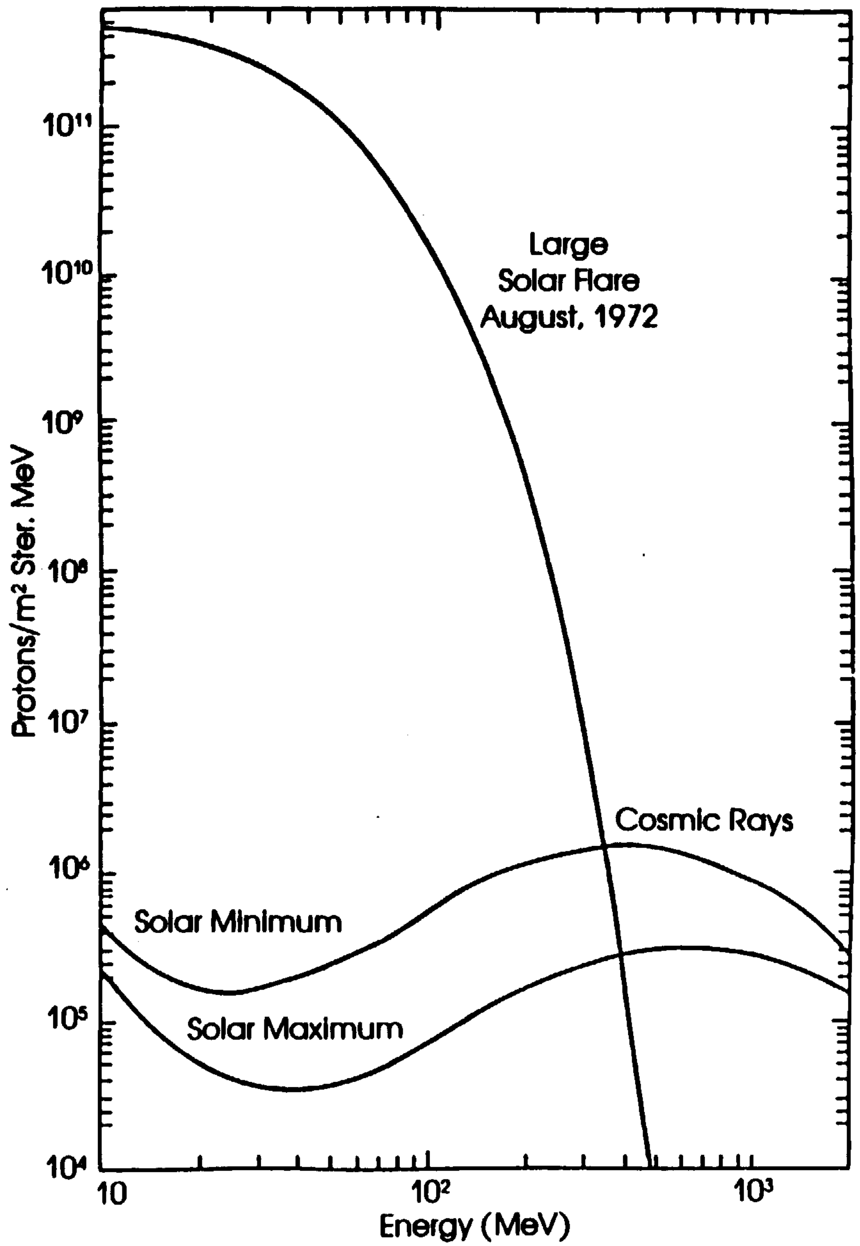
From "Proton Events During the Past Three Solar Cycles", Smart, D.F., and Shea, M.A.

Proton Energy Spectrum

This chart provides a comparison of the time-integrated spectrum for the solar proton event of August, 1972 with the galactic cosmic ray proton spectra accumulated in one week during solar minimum and maximum. The spectral distribution at Earth changes as a function of time because high energy particles tend to arrive before those with lower energy. The angular distribution of the particles also varies from event to event. During some of the high energy events the particles tend to be directional early in the event. The arrival of the lower energy particles tends to be more isotropic in nature.

Proton Energy Spectrum



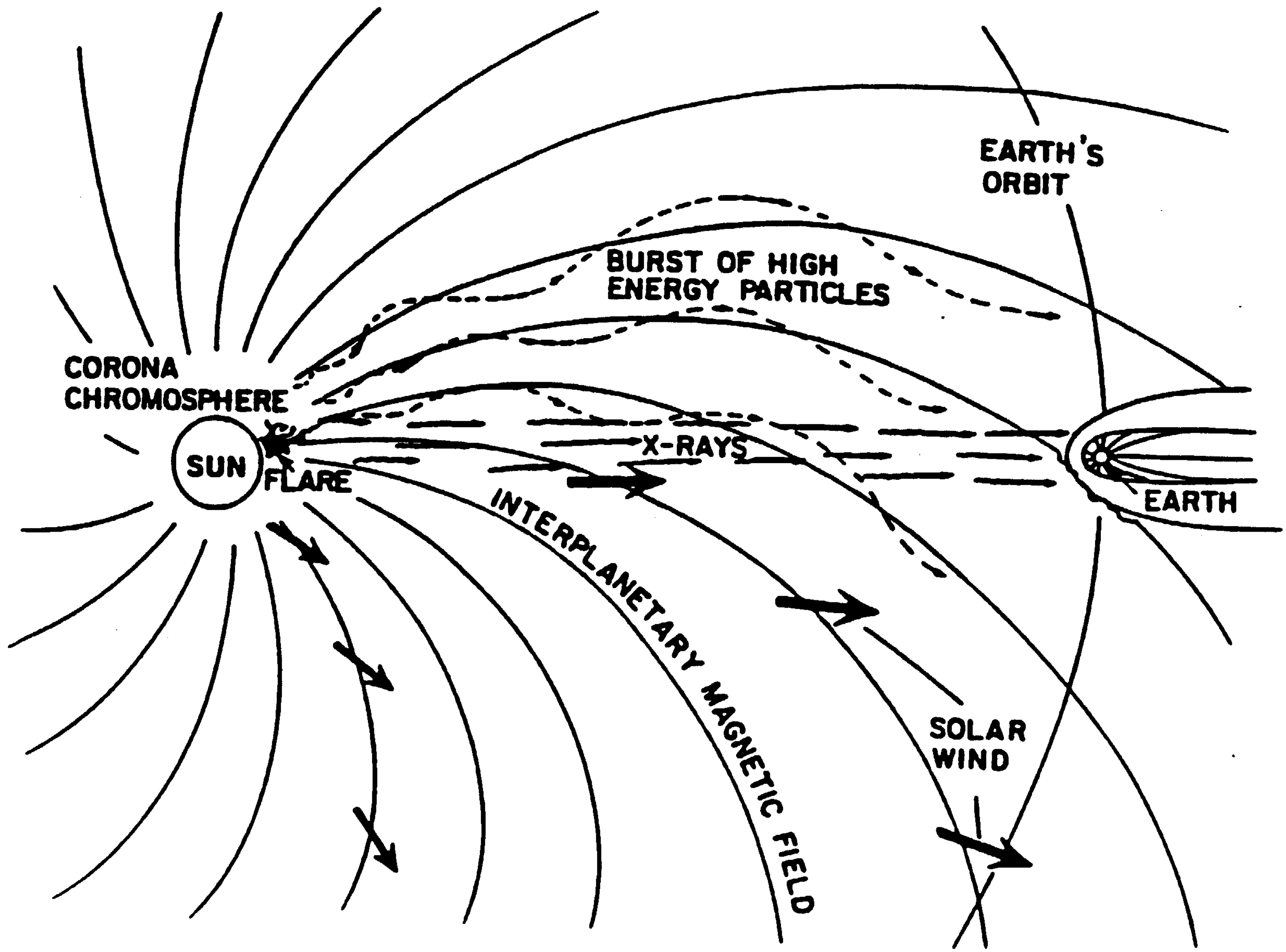
* From the SICSA Outreach Journal; Vol. 2,No. 3, July-September, 1989, J.R. Letaw, R. Silberberg and C.H. Tsao

Characteristics of the Idealized Structure of the Interplanetary Medium

Unlike solar electromagnetic radiation, both the onset time and the maximum intensity of the solar particle flux depend on the heliolongitude of the flare with respect to the detection location in space. The directionality results because particles will move more easily along the interplanetary magnetic field direction. The interplanetary magnetic field topology is determined by the solar wind outflow and the rotation of the sun which during "quiet" conditions can be approximated by an Archimedian spiral shown in the figure.

The charged particles emitted during a solar flare consists of a nonequilibrium plasma cloud which expands to several solar diameters as it migrates away from the Sun. The particle fluxes observed by a detector inside this plasma cloud are essentially isotropic and these particles constitute the larger portion of the total flare radiation.

Characteristics of the Idealized Structure of the Interplanetary Medium

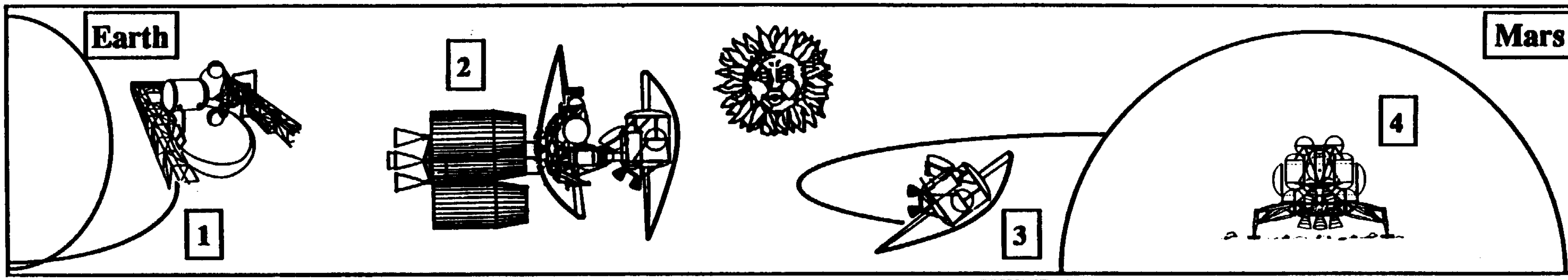


From "Proton Events During the Past Three Solar Cycles", Smart, D.F., and Shea, M.A.

Radiation Environments for Mars Mission Phases

This chart describes radiation environments which are of most concern to mission planners and vehicle configurators. The Mars exploration class mission has been divided into four phases; (1) on-orbit assembly and checkout, (2) vehicle transfers, (3) the Mars orbital sequence, and (4) surface stay time referenced at 30 days. Galactic cosmic radiation and the highly unpredictable (time of occurrence and magnitude of event) solar proton events constitute the overwhelming threat to crew and vehicle except in those areas that fall under the protective coverage of the magnetosphere. In this regime the trapped radiation (Van Allen Belts) and in particular the South Atlantic Anomaly (SAA) pose the largest concern during vehicle assembly and checkout. During the surface exploration phase of the mission additional protection is provided by the Mars atmosphere. The environments are variable both temporally and spatially. This variability occurs for a number of reasons including fluctuations in solar activity associated with the solar cycle, altitude and inclination in LEO, mass of the planetary body, and diurnal and seasonal variations which influences atmospheric density on Mars and consequently changes its attenuating properties. Natural and man-made shielding that will be influencing design work for each of the phases has also been listed in the final column.

Radiation Environments for Mars Mission Phases



Reference	Mission Phase	Environment	Variability	Shielding
1	On Orbit Assembly and Checkout	Trapped Radiation SAA	Solar Activity Altitude, Inclination Earth Proximity	• Geomagnetic Field • Vehicle Structure, fluids, stores equipment and spares
2	Vehicle Transfer Earth/Mars/Earth	Trapped Radiation GCR, SPE	Solar Activity	• Interplanetary magnetic field • Vehicle structure, fluids, stores equipment, spares, & waste • Vehicle orientation
3	Mars Orbit	GCR, SPE	Solar Activity Mars Proximity	• Planet mass • Vehicle structure, fluids, stores equipment, spares, & waste • Vehicle orientation
4	Mars Surface	GCR, SPE	Solar Activity Diurnal and Seasonal Variations	• Mars atmosphere • Planet mass • Mars surface material • Vehicle structure, fluids, stores equipment, spares • Deployable "zenith" shield

Mission Opportunity Stay Time Coincidence with Predicted Solar Maximum and Minimum Years

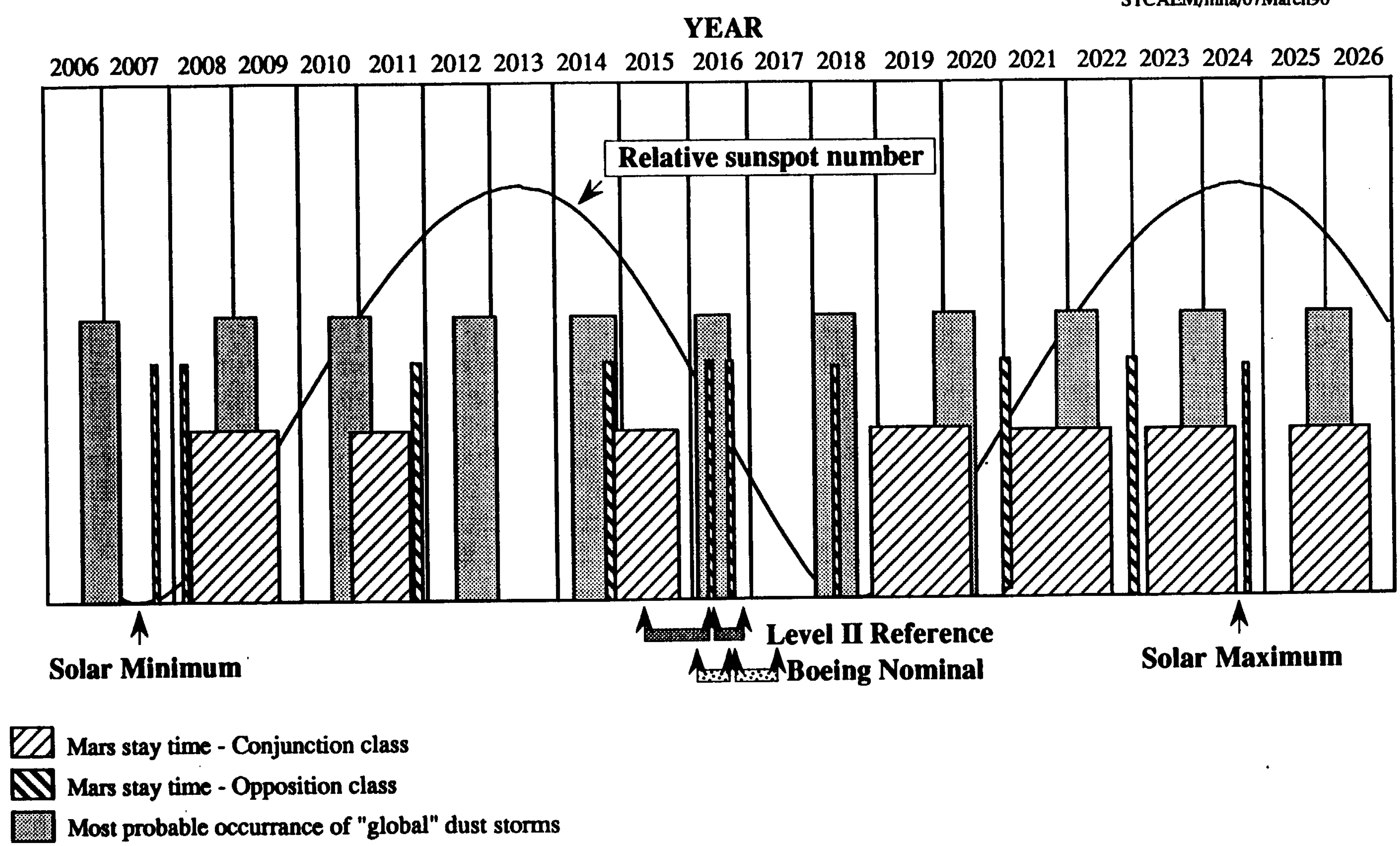
As expected solar flares are often the greatest source of radiation dosage received on a long duration mission outside the protective shield of the Earth's magnetosphere. The timing of a mission can have a major effect upon the expected dose. In addition to showing Mars stay times and the most probable occurrences of "global dust storms" a curve representing the relative sunspot number is shown. Records of sunspots have been kept for over two centuries. The 11 year cycle of sunspots is only approximated but the cyclic behavior is unmistakable.

During solar maximum, when the interplanetary magnetic field strength is greatest, cosmic ray particles are attenuated more effectively producing a GCR flux minimum (the Fornbush decrease), conversely, GCR flux is largest during solar minimum.

Solar proton events change in frequency and size during the 11 year sunspot cycle, reaching maximums before and after sunspot maximum. This chart shows that greater concern for occurrences of SPE's is not to be directed only at those years of predicted solar maximums but also in the regions on the curve surrounding the solar maximum.

Mission Opportunity Stay Time Coincidence With Predicted Solar Maximum and Minimum Years

STCAEM/mha/07March90



Dose Equivalents to the BFO for Various Mission Phases to Mars with Representative SPE's

This chart is meant to provide reference data to compliment the chart titled "Radiation Environments for Mars Mission Phases". A representative opposition mission with total transit time of 430 days and stay time of 30 days on the Martian surface was used to determine the respective dose equivalents to the blood forming organs during the various mission phases. The black bars show these dosages and also indicate the duration in which the crew remain in the particular environment. In addition to these "constant" forms of radiation three representative flares are also presented to show the potential hazard of these unpredictable events. The August 1972 event occurred toward the end of solar cycle 20 previously thought to be "stable". Prior to this event the 1956 and 1960 events shown were described as the worst recorded cases. It is important to note that the 1972 flare could have been worse if it had occurred at a more "favorable" position on the sun relative to the Earth. The chart clearly shows the immense dosage that can be received during such a short duration event. The data on this chart assumes the protection to the crew would come from 0.77cm of aluminum shielding except when the crew is on the surface in which the Martian atmosphere adds an additional 3.85cm (aluminum) of effective shielding. Another important point to note about the chart is that the 1956 flare was more energetic than the 1972 flare. The high fluence associated with the 1972 flare and longer duration give it greater "ionizing power". In order to reduce the received dosage below the 30 day limit an effective net shielding would have and areal density of ~24g/cm^2 or shield thickness of ~8.9cm.

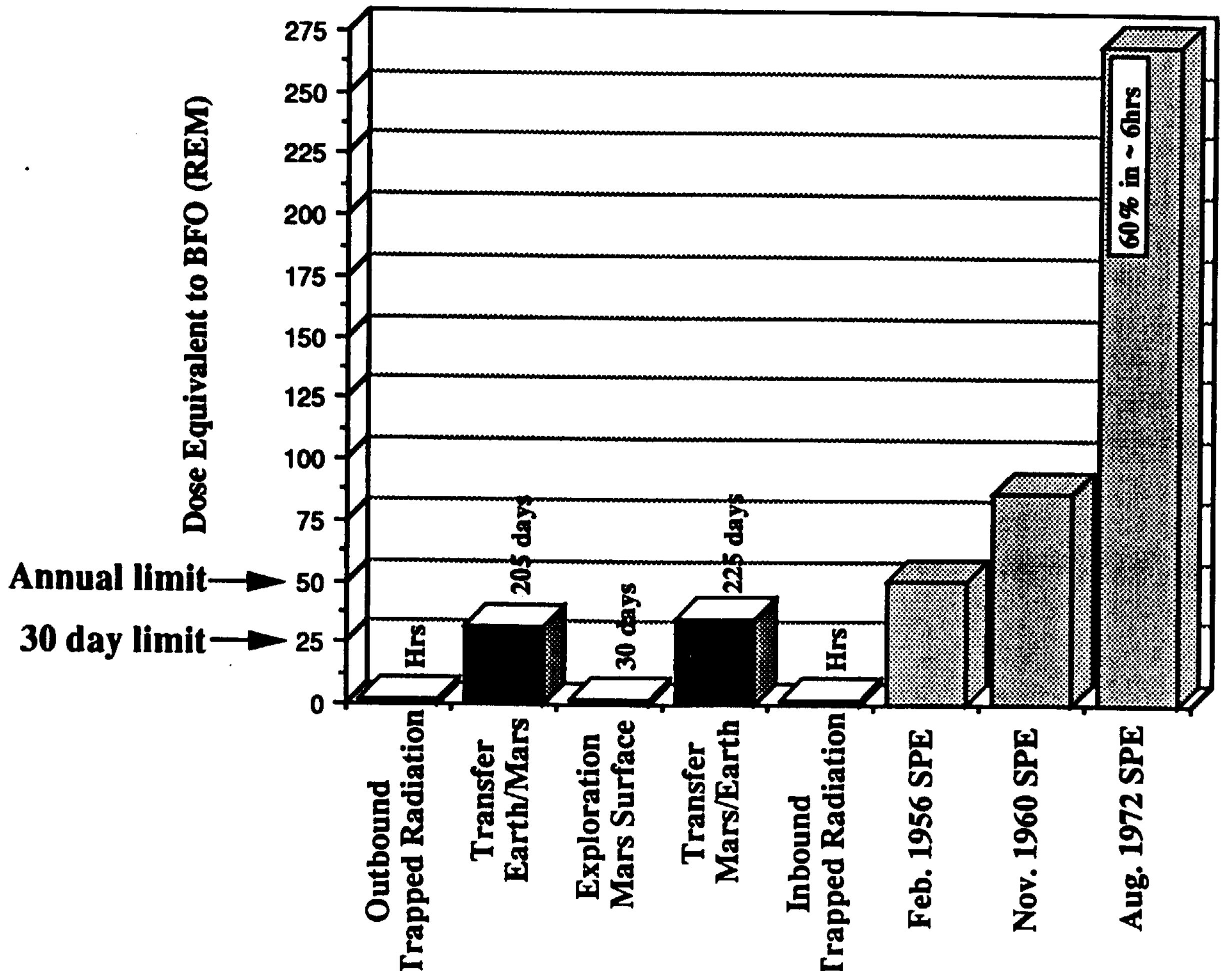
ADVANCED CIVIL SPACE SYSTEMS

Dose Equivalents to BFO for Various Mission Phases to Mars with Representative SPEs

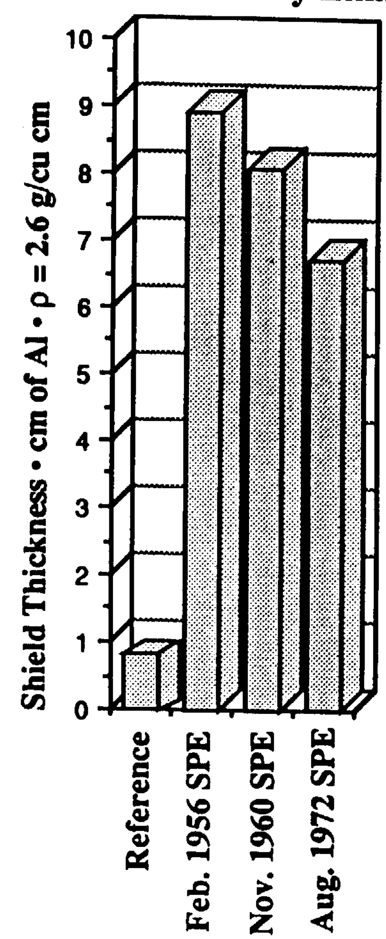
BOEING

STCAEM/mha/07March90

- All shielding assumed to be 2 g/sq cm except for Mars surface where additional 10 g/sq cm for atmosphere



Thickness to Reduce Below 30 Day Limit



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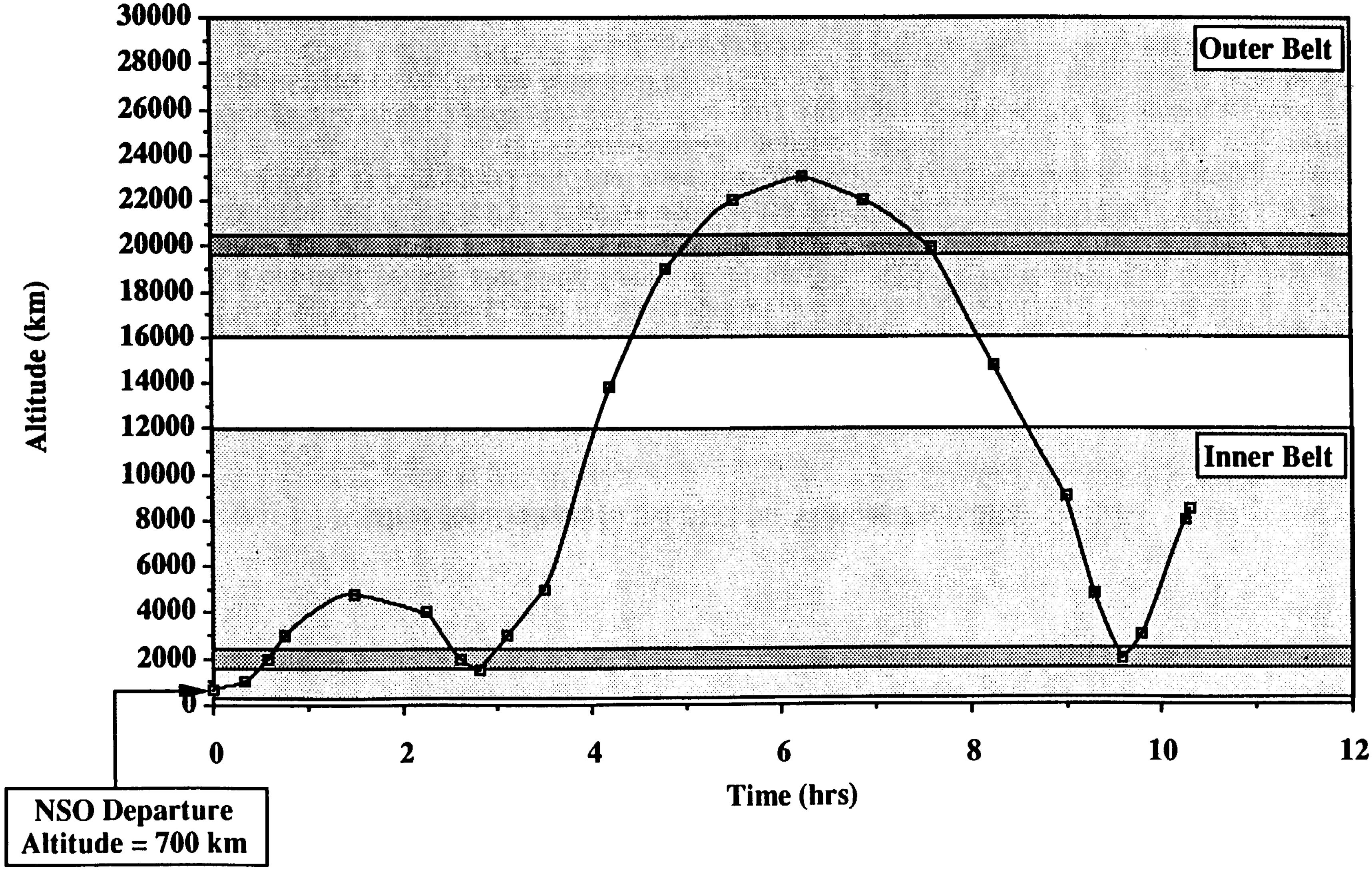
Data from S. Nachtwey, JSC, NASA and J.E. Nealy, Langley, NASA
D615-13005

Low Thrust NTR - 3 Burn TMI - Altitude vs. Time

Radiation exposure times for multiple burn trans-Mars injections are inherently higher than those using single burn strategies. If our low thrust NTR vehicle departs from a nuclear safe orbit (NSO) of 700km and orbital inclination of 28.5 degrees, the total elapsed time to perform the three burns through the inner and outer belts will be approximately 9.75 hours. The upper altitude, lower altitude, and peak regions (indicated by the thin darker band) of the belts vary with solar activity. Conservative values for these altitudes based on literature research have been selected to allow determination of radiation exposure to crew members under .75cm (2g/sq cm) of aluminum. This altitude vs. time plot used in conjunction with the JSC AP-8 and AEI-7 codes were used to determine an approximate dose to the blood forming organs. Preliminary analysis indicated that crew members would receive on the order of 4 rems, not a significant amount but much higher than that received in a straight passage through the belts as the Apollo lunar missions.

Low Thrust NTR - 3 Burn TMI - Altitude vs. Time

- Altitude parameters of inner and outer radiation belts will vary with solar activity and changes in latitude



Dose Equivalents to the BFO for Various Propulsion Options

It is possible to predict the amount of ionizing radiation a crew member will receive from the "constant" forms of radiation such as the trapped and galactic cosmic radiation. The exposure to the blood forming organs (assumed to be the limiting system for total dose) will be determined from the bodies own shielding capabilities, the amount of bulk vehicle shielding, total time of exposure,and the energy associated with the charged particles. If we assume a constant vehicle shield thickness of .75cm ($2 g/cm^2$), a Mars stay time of 30 days under a conservative areal density of 10 g/cm^2 , and various trip times through the trapped radiation belts (propulsion option and mission profile dependent), the following dosages would result. Crew members will not be on board SEP and NEP vehicles as they spiral out from a nuclear safe orbit. Transportation to the slowly accelerating vehicle will be accomplished by an OTV, consequently radiation exposures to crew members will constitute a single pass through the trapped radiation belts. The only variation shown to the accumulated dose passing through the trapped belts comes on the outbound leg of the low thrust NTR. The value shown here reflects the use of the previous chart and the JSC AP-8 and AEI-7 codes. Radiation exposures do not include that which may come from nuclear propulsion options or that incurred from solar proton events.

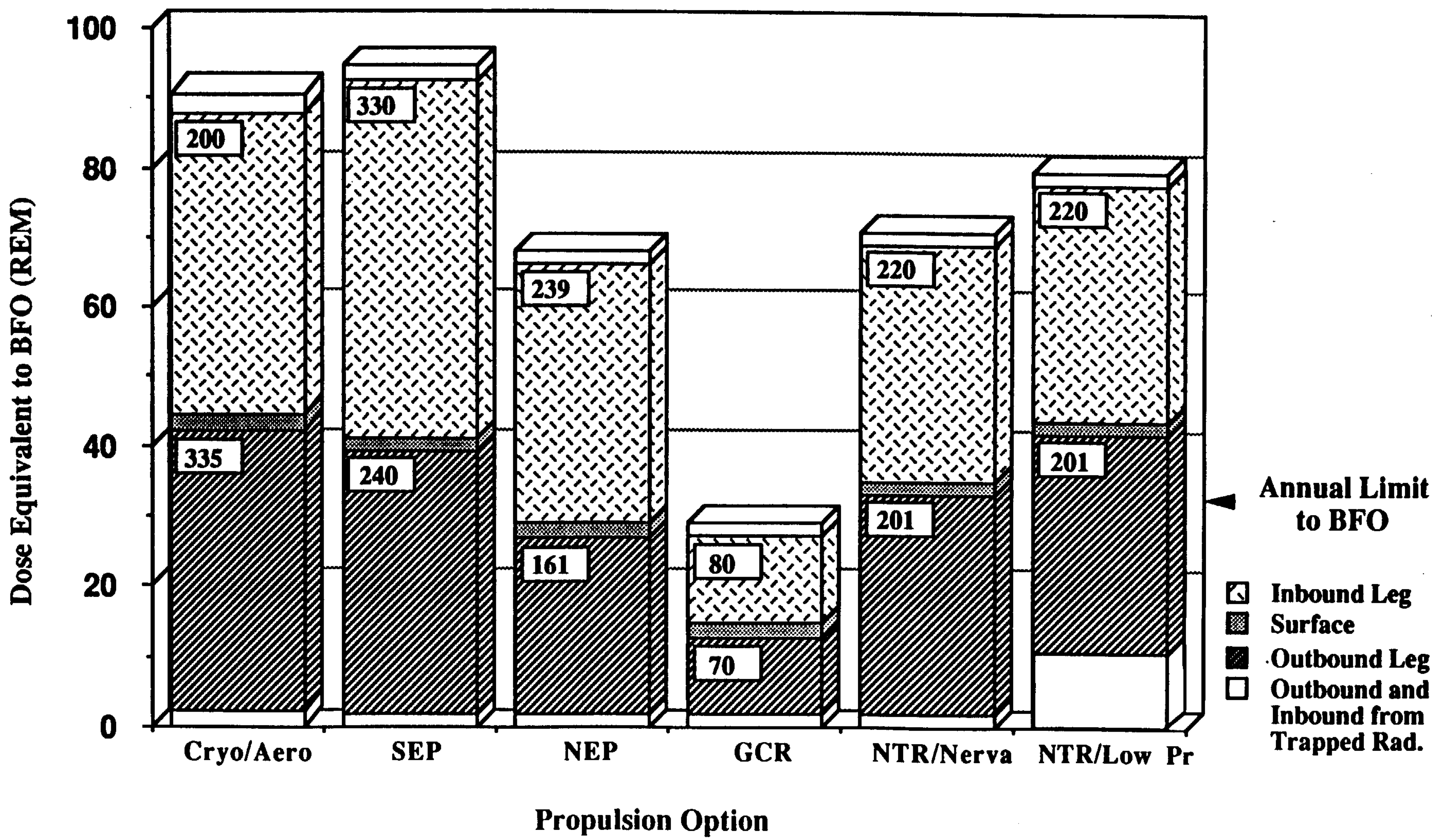
ADVANCED CIVIL SPACE SYSTEMS

Dose Equivalents to BFO for Various Propulsion Options

BOEING

STCAEM/mha/07Feb90

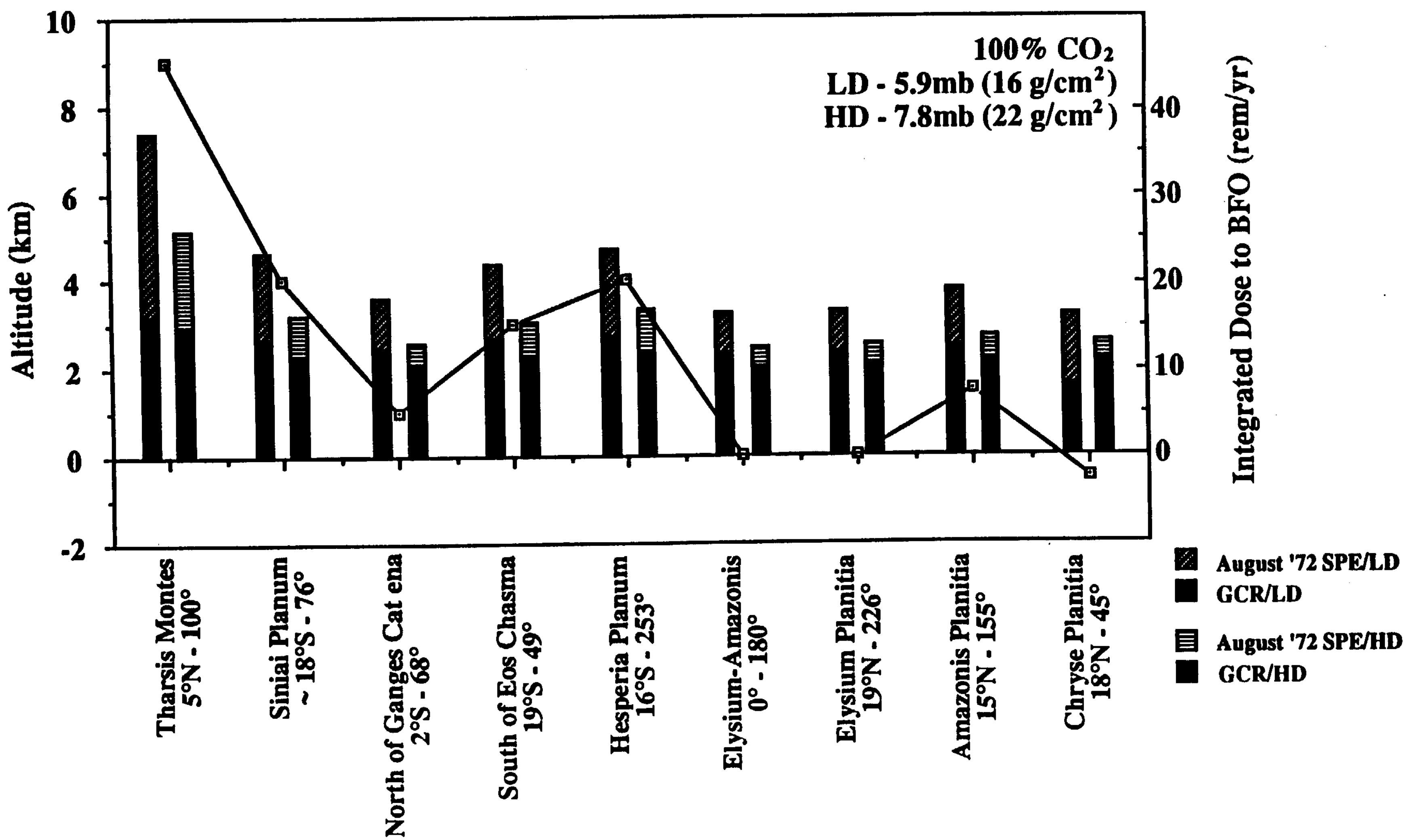
• All shielding assumed to be 2 g/sg cm except for Mars surface where additional 10 g/sq cm for atmosphere
• Stay time on Mars - 30 days



Altitude and Dose Comparison for Mars Using High and Low Density Atmospheric Models

One major concern to mission planners and vehicle designers will be the damaging effects of ionizing radiation from high energy galactic cosmic radiation (GCR) and solar proton events. Crew members will encounter the most harmful radiation exposures during the transit phase of the mission. Once on the surface the tenuous Martian atmosphere should provide significant protection from the harmful radiative fluxes. Variations in the amount of this protection will be the result of changes in the altitude, pressure (seasonal), and the angle from the zenith of the incoming high energy particles. This chart indicates two of those variations, altitude and pressure. Assuming that the composition of the Martian atmosphere is one-hundred percent CO_2 (actually ~ 95%), high density (HD) and a low density (LD) models were used to determine the effective shielding provided by the atmosphere. As the pressure increases so to does the potential shielding. In addition to the continuous radiation coming from the GCR flux , one large representative solar flare (August 1972) was added to the integrated GCR exposure over one year to give the annual dose to the blood forming organs. These models assume that incident particles are coming from straight overhead. The chart shows the amount of radiation that would be received at various potential landing sites. As one would expect the greater the altitude of the site, the greater the exposure. The low and high density models indicate the variations that may be encountered with changing season and the movement of the CO_2 to the polar regions. The line graph indicates the relative variations in the altitude.

Altitude and Dose Comparison for Mars Using High and Low Density Atmospheric Models



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Ionizing Radiation Protection Design Considerations

Four primary protection methodologies considered

- **Bulk mass shielding**
- **Active electromagnetic shielding**
- **Use of chemical protectors**
- **Avoidance of high radiation fluxes**

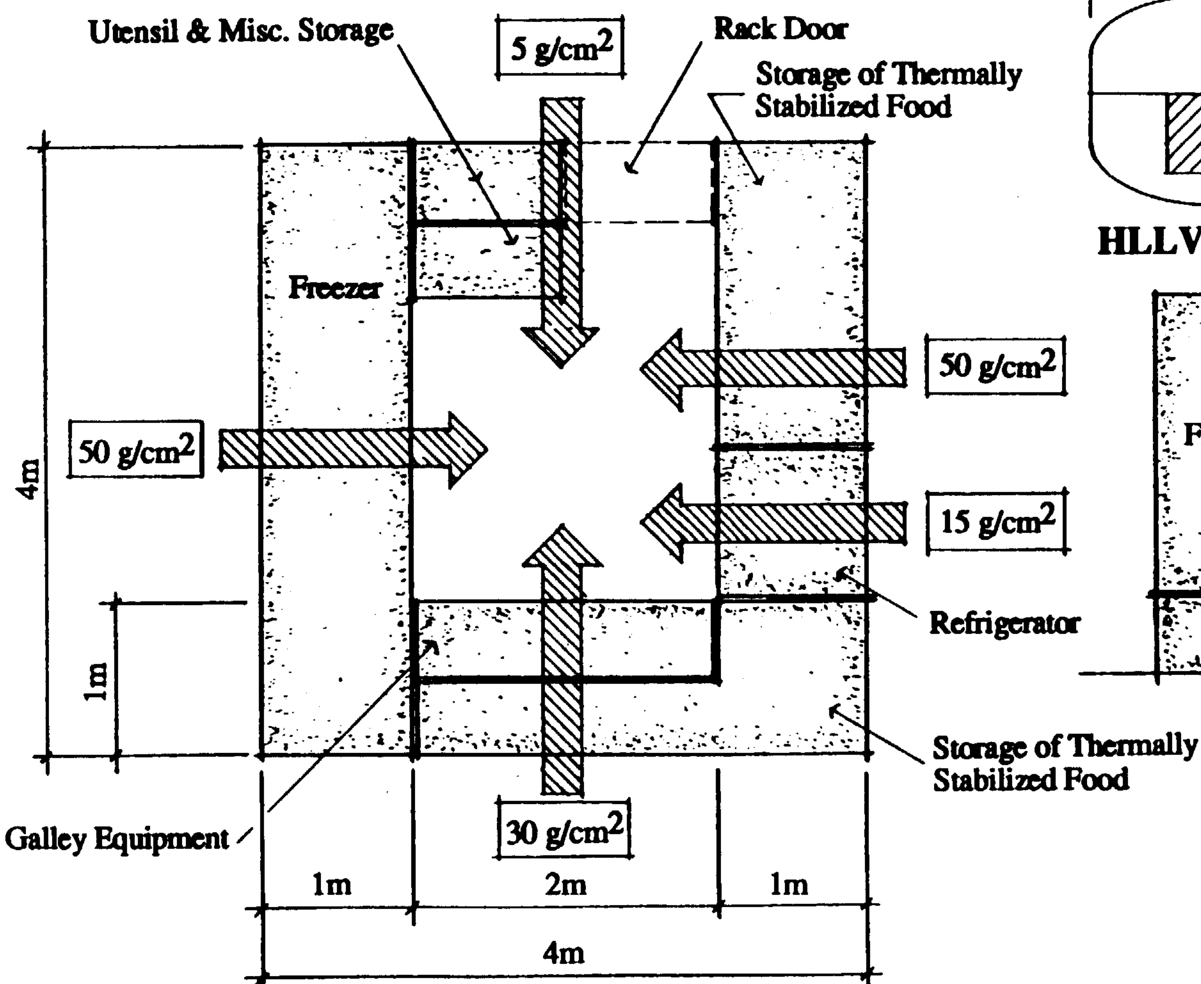
MTV Habitat Galley/Storm Shelter

A primary design consideration for radiation protection is the use of bulk shielding. One novel concept now being evaluated is the use of a storm shelter/galley configuration. The next four charts show a plan and section view of this concept and then provide back drop information. As a first order approximation areal densities are extremely good but do not provide 4 pi protection. It will be necessary to explore further the use of "composite" walls and innovative means of packaging and storing equipment and consumables. Analyses of various potential protection concepts will be key upcoming work.

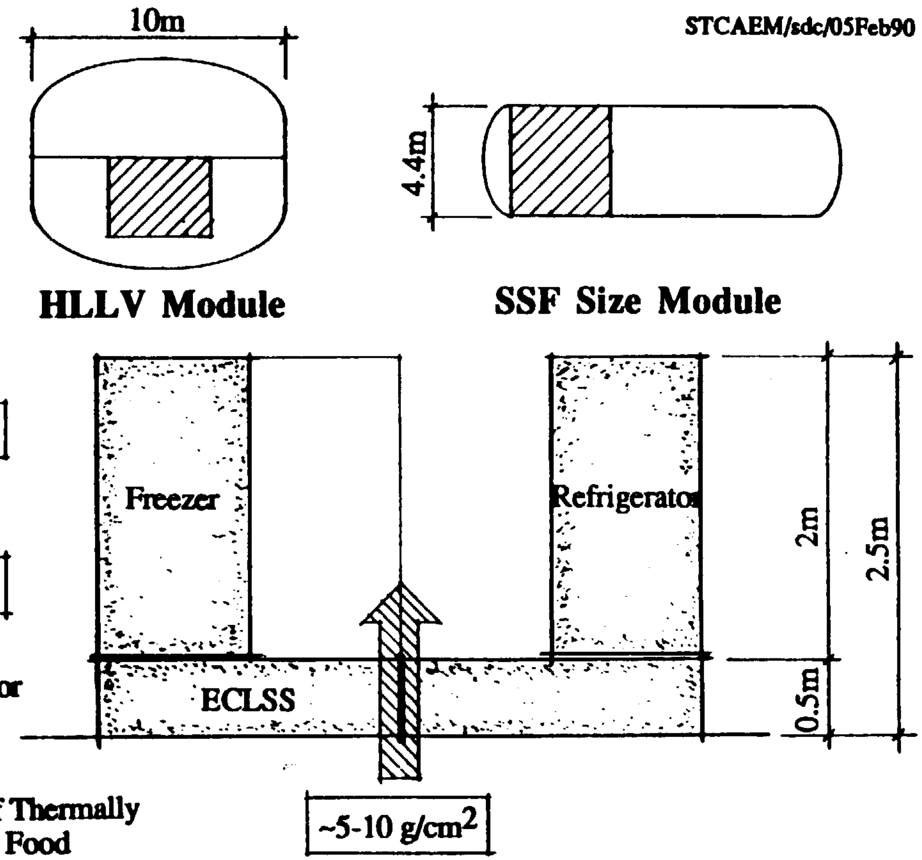
MTV Habitat Galley/Storm Shelter

BOEING

STCAEM/sdc/05Feb90



Plan



Section

- 40m3 total volume
- 9m3 free volume
- Floor and ceiling require dedicated composite shield panels.

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Consumables Provisioning for MTV

- **Stored food** (2.0 kg/crew/d brought from Earth)

  0.62 food solids
  1.15 65 % water in wet food (water "surplus")
  0.23 packaging

- **Storage density**

  0.6 t/m3 frozen or thermally stabilized
  0.2 t/m3 fresh

- **Potable water** (2.35 kg/crew/d provided recycled by ECLSS)

  1.59 drinking
  0.76 food prep

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MTV Food

Mission constraints

- Maximum mission time 1020 d

- Food preparation and consumption is one of the most critical means available to boost morale and stabilize groups in hazardous, long-duration confinement.

Derived requirements

- At least SSF quality; some actual cooking advisable

- 5 % fresh (controlled atmosphere storage; 1 yr lifetime possible)
  50 % frozen (limited-access deep freeze)
  40 % thermally stabilized
  5 % dried (beverages, soups)

- 3 % supplemental may be grown onboard (not mission-critical)

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Synergistic Usage of Consumables

With wet food, advanced water recovery is not required

- Avoids development cost, operational risks of high-energy water systems
- SSF ECLSS with enhanced long-duration reliability is satisfactory for MTV

Consumables are valuable for radiation shielding

- 8.2 t of packaged food available on a 4 crew, 1020 d mission
- Only 530 kg is unrecoverable with SSF ECLSS (fecal solids and water)
- Brine requires minimal biological stabilization
- Food packages stored in blocks; empty blocks become brine containers, filled by ECLSS; manually replaced into storage frame; shield wall continually maintained throughout mission

Combined galley / storm shelter reduces shielding penalties

- Dramatically limits dedicated shielding mass otherwise required
- Temporary sleep accommodations rigged for flare duration
- Separate shelter provisioning not required

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Radiation Research Concerns

- Reevaluating the tradeoff between simpler radiation schemes and potentially increased career cancer risks
- Reevaluation of the "conventional" risk assessment now being used
- Development of SPE and dosimetry warning systems
- Trade studies required for realistically selecting and assessing shielding questions such as material, mass, size, and structural integrity
- Evaluate the potential of exacerbating the effects of radiation with a weightless or reduced 'g' environment
- Effectiveness of chemical inhibitors and nutritional supplements
- Evaluation of shielding technologies including: waste water, lightweight composite materials, electromagnetic shielding and propellants
- Analysis of trajectories that may come as close as 0.6 AU to the sun

LifeSat

In terms of the external and internal radiation environments, it will be essential to obtain further data and reliable descriptions of the fluxes and types of primary and secondary particles. The Life Sciences Division is currently planning a reusable, free-flying biological satellite program (LifeSat), that will provide the capability to study the biological effect of radiation dosages and the effectiveness of various shielding materials. Accurate information will be provided on a unique spectrum of radiaton that will be extremly valuable for risk assessment and protection methodology. It has been estimated that a 60-day mission in polar orbit would simulate 5% of a Mars mission in terms of radiation exposure.

LifeSat

Goal:

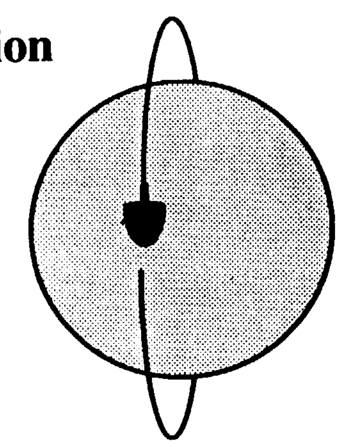
- Understand the effects of microgravity and radiation in the space environment through the capabilities of a reusable free-flying satellite

Objectives:

- Provide accessibility to range of orbits, including polar
- Provide long-duration missions of approximately 30-60 days
- Provide capability to perform research at artificial gravity levels between 0 and 1.5 g

Status:

- Recently awarded Phase B contracts
- Budget estimates assume significant international collaboration
- Schedule support potential FY 1992 New Start



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Boeing Huntsville IR&D Transport Code Development Objectives

- Research codes available to model primary and secondary radiation effects
- Obtain codes satisfying engineering requirements of various environments and mission profiles
- Become code proficient and generate subroutines that will model a variety of radiation environments
- Develop subroutine to model variety of materials and vehicle geometries
- Code verification
- Document code modifications in users guide

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Aerobrake Mass Analysis

Brent Sherwood

Agenda

Design Assumptions
Aerobrake Structural Design
Structural Mass Comparison

Aerobrake Structural Design
Design Assumptions

A structural design study was undertaken to determine reasonable weights for rigid aerobrakes. Listed here are the primary assumptions for the study, based on mission analysis requirements, configuration constraints, and previous aerobrake structure investigations.

Aerobrake Structural Design

Design Assumptions:

(1) Constant spar cross-sections, curved profiles

(2) C/Mg metal matrix spars (Density 1830 kg/cu. m.)

(3) Payload: Mars Excursion Vehicle, 81MT
Mars Transfer Vehicle, 153MT

(4) 6g maximum acceleration

(5) 8 payload attach points (4 frame and 4 landing leg points)

(6) Relative Wind Angle = 20 degrees

(7) Variable pressure distribution, Range 1.5psi to 3psi (81MT); 2.9psi to 5.9psi (153MT)

(8) Structure temperature = 394K (250F)

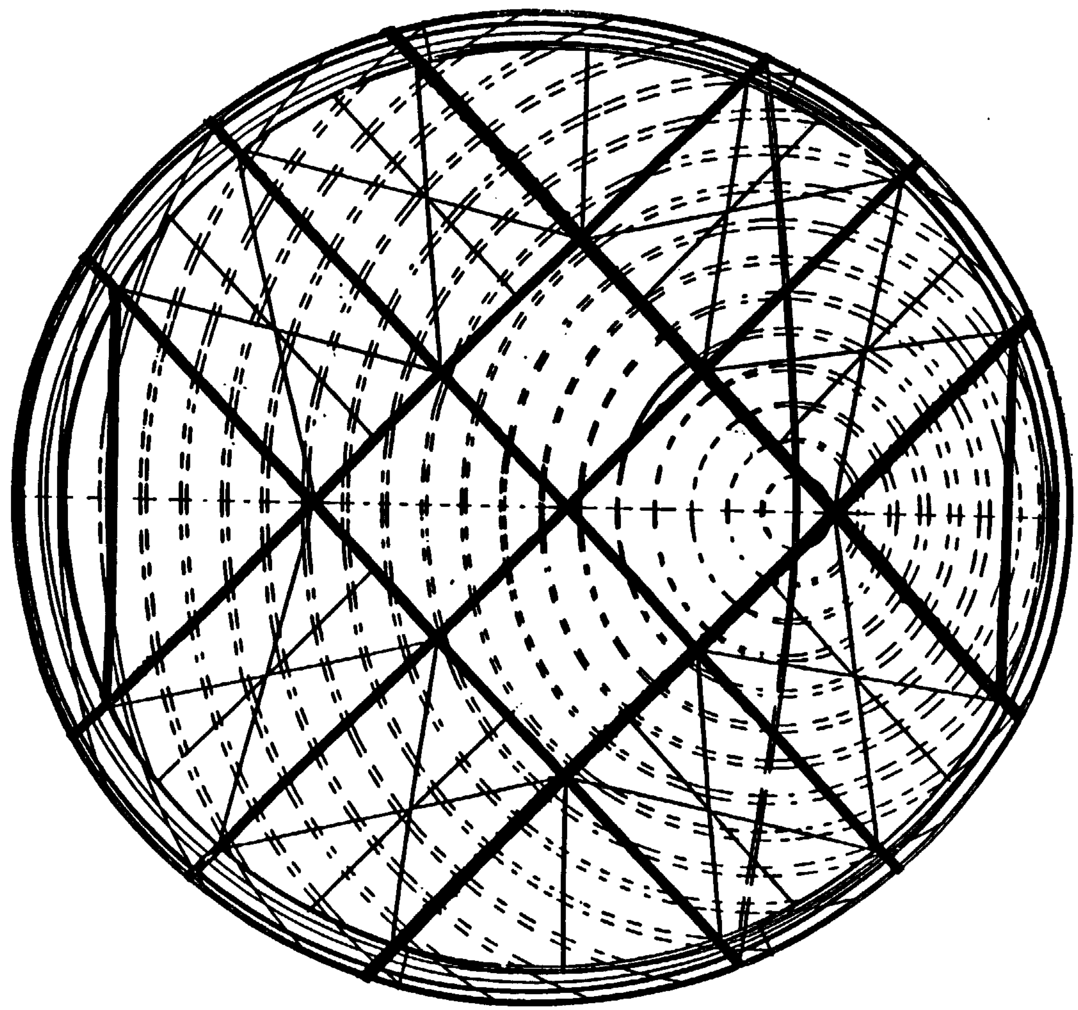
(9) Secondary spar pattern to be triangular for greater shear resistance

Aerobrake Structural Design
Plan View of Aerobrake Structure

This layout shows the primary and secondary spar pattern, organized around the off-center, square aerobrake port pattern required for the MEV descent engines. Secondary spars contribute triangulation. The brake is assumed to come apart in nine separate pieces for packaging in an ETO launch shroud.

Aerobrake Structural Design

Plan View of Aerobrake Structure:



Aerobrake Structural Design
Materials

Assumed materials and their performance characteristics are shown.

Aerobrake Structural Design

Materials:

Primary and secondary spars: Cast C/Mg metal matrix	Ultimate Tensile Strength = 459MPa (66.5 ksi) Density = 1830 kg/cu. m. (.066 lb/cu. in.)
Ti - 6Al - 4V face sheets:	Ultimate Tensile Strength = 152 ksi @ 250F Yield Compressive Strength = 143 ksi Ultimate Shear strength = 93 ksi @ 250F Density = 4430 kg/cu m (0.16 lb/cu. in.)

Honeycomb Core:

Al 5052 Flexible, BMS 4-6, Type 4.1-25: (chosen to accommodate curvature)	Density = 4.1 lb/ cu. ft.

Aerobrake Structural Design
Results

Component (by structure subsystem) and total aerobrake mass estimates are shown, assuming two different reference spar depths and two different spar strengths. The first chart of the pair shows results for the 81mt payload of the reference MEV. The second shows results for the 153 mt payload of the reference MTV.

Aerobrake Structural Design
81 mt payload

Results (Weights assuming 81,000 kg payload, MEV):

19.5 inch spar depths:	105 ksi spar strength		200 ksi spar strength	
Primary spar weight:	5,390 kg	(11,859 lb)	2,751 kg	(6,052 lb)
Secondary spar wt:	3,827 kg	(8,420 lb)	2,975 kg	(6,546 lb)
Honeycomb weight:	6,758 kg	(14,868 lb)	6,758 kg	(14,868 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	19,275 kg	(42,407 lb)	15,784 kg	(34,726 lb)
22.5 inch spar depth:				
Primary spar weight:	4,989 kg	(10,978 lb)	2,484 kg	(5,465 lb)
Secondary spar wt:	3,809 kg	(8,379 lb)	2,596 kg	(5,711 lb)
Honeycomb weight:	6,758 kg	(14,868 lb)	6,758 kg	(14,868 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	18,856 kg	(41,483 lb)	15,138 kg	(34,726 lb)

Note: 200 ksi option may require additional material technology development efforts.

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Aerobrake Structural Design

Results (Weights assuming 153,000 kg payload, MTV):

19.5 inch spar depths:	105 ksi spar strength		200 ksi spar strength	
Primary spar weight:	10,313 kg	(22,689 lb)	4,816 kg	(10,596 lb)
Secondary spar wt:	6,296 kg	(13,851 lb)	3,749 kg	(8,248 lb)
Honeycomb weight:	12,785 kg	(28,127 lb)	12,785 kg	(28,127 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	32,694 kg	(71,927 lb)	24,650 kg	(54,230 lb)

22.5 inch spar depth:				
Primary spar weight:	8,671 kg	(19,076 lb)	4,239 kg	(9,327 lb)
Secondary spar wt:	5,381 kg	(11,838 lb)	3,434 kg	(7,555 lb)
Honeycomb weight:	12,785 kg	(28,127 lb)	12,785 kg	(28,127 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	30,137 kg	(66,301 lb)	23,758 kg	(52,267 lb)

Note: 200 ksi option may require additional material technology development efforts.

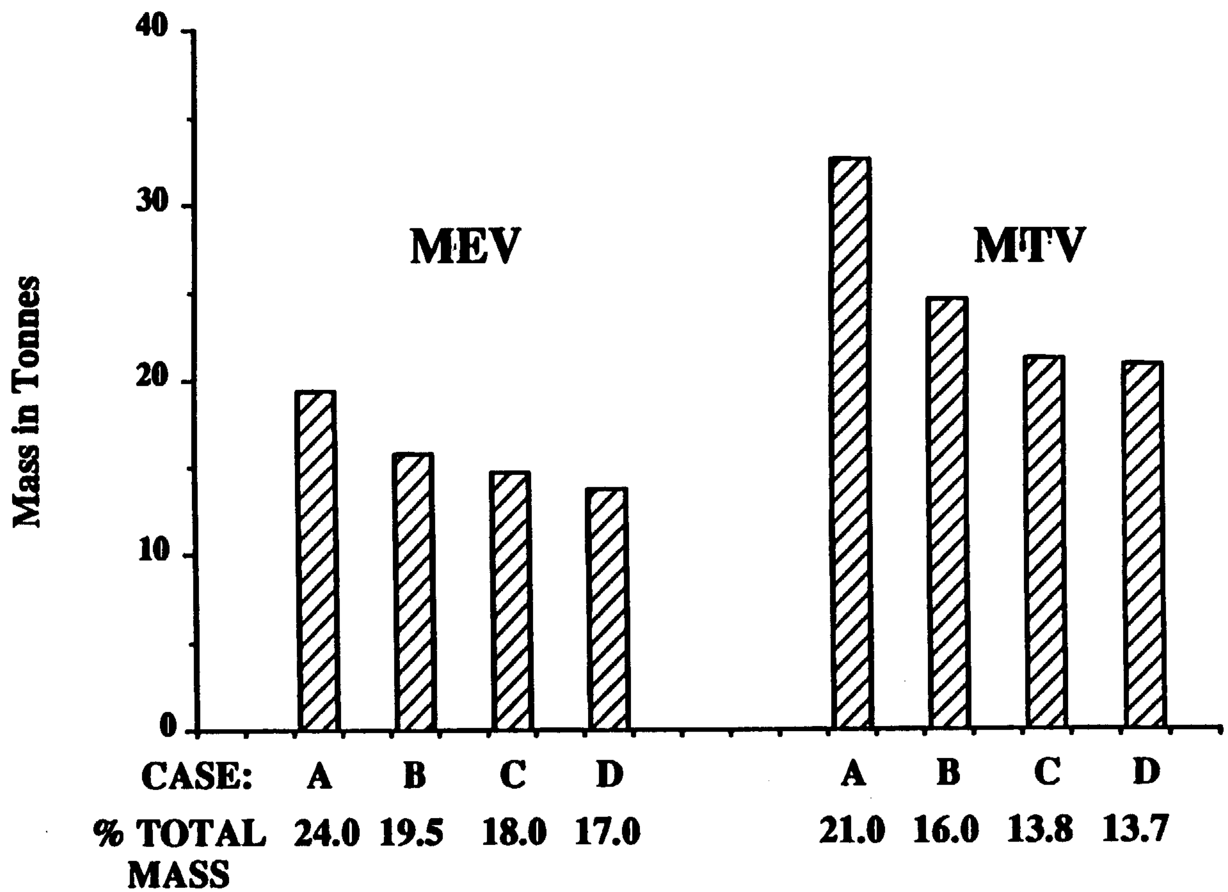
Mars Aerobrake Structural Mass Comparison

Compared in the figure are the total aerobrake mass for the combinations of materials and loading options indicated, for both MEV and MTV reference payloads, for the L/D = 0.5 rigid aerobrake. Such aerobrakes can be expected to comprise between about 14 - 19 % of their total captured payload. Aerobrake mass does not scale simply with payload. Flying lower-g trajectories both lowers the loading and the temperature (hence the TPS mass) and reduces the total aerobrake mass.

Taking advantage of so-called "deep structure" concepts and integrated TPS/facesheet materials concepts holds the potential to reduce rigid aerobrake mass further.

Mars Aerobrake Structural Mass Comparison

BOEING



0.5 L/D aerobrake

Cases:	Spar strength	Honeycomb	g level
A:	105	Ti	6
B:	200	Ti	6
C:	200	GrEp	6
D:	200	Ti	5

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Shuttle-Z, TMIS

ADVANCED
CIVIL
SPACE
SYSTEMS

Agenda

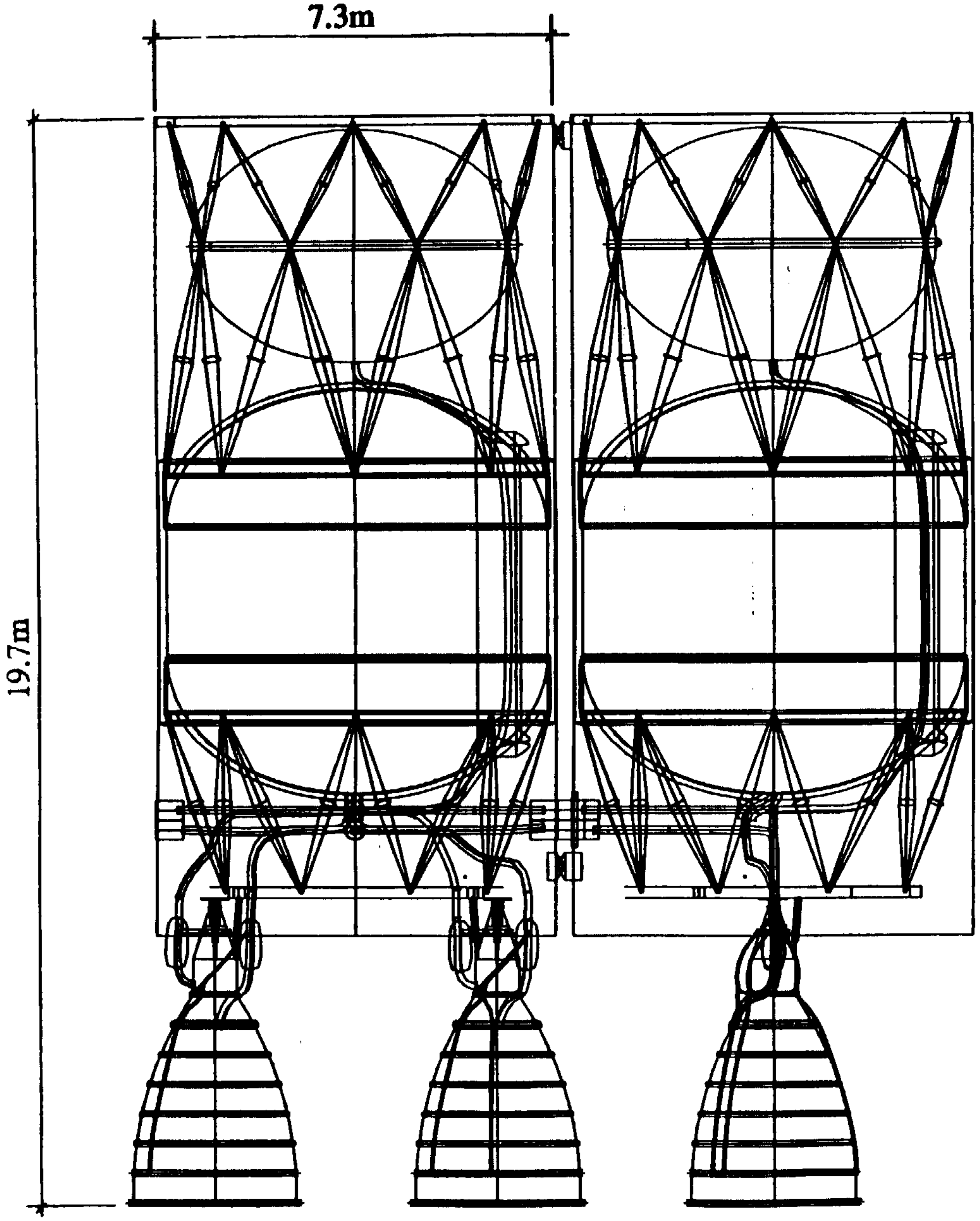
Ten Engine Option
Five Engine Option

Shuttle-Z 3rd Stage TMIS

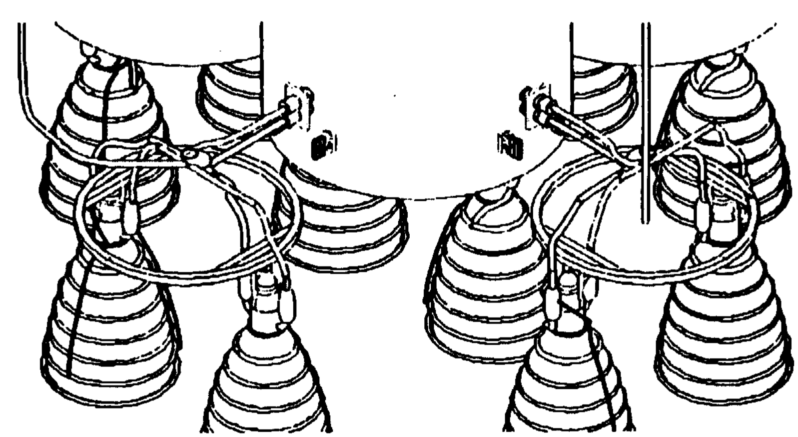
A concept was developed for a Trans-Mars Injection Stage comprised of clustered upper stages from the MSFC Shuttle-Z ETO concept. The initial stage concept unit had a propellant capacity as shown, with engine-out capability for each unit on ETO ascent. TMIS construction would be accomplished by automated rendezvous and docking (AR&D), using a hinged first-contact-point latch at the forward end. Fluid, data and power connections would be made subsequently and automatically, upon verification of successful structural attachment, using a separately actuated interconnect fitting. Cross-manifolding would then allow engine-out burns on TMI.

The next two charts show a 3-D model of the concept.

"Shuttle-Z 3rd Stage" TMIS



- Modular TMIS accommodates 100-508 t propellant.
- "Fly-together" automated rendezvous & docking.
- Core unit: - Umbilical connections for up to 4 strap-on units.
  - Contains interconnect manifolds.
- Strap-on units: - All identical.
  - (2) 150 klbf engines.
  - 102 t propellant maximum.



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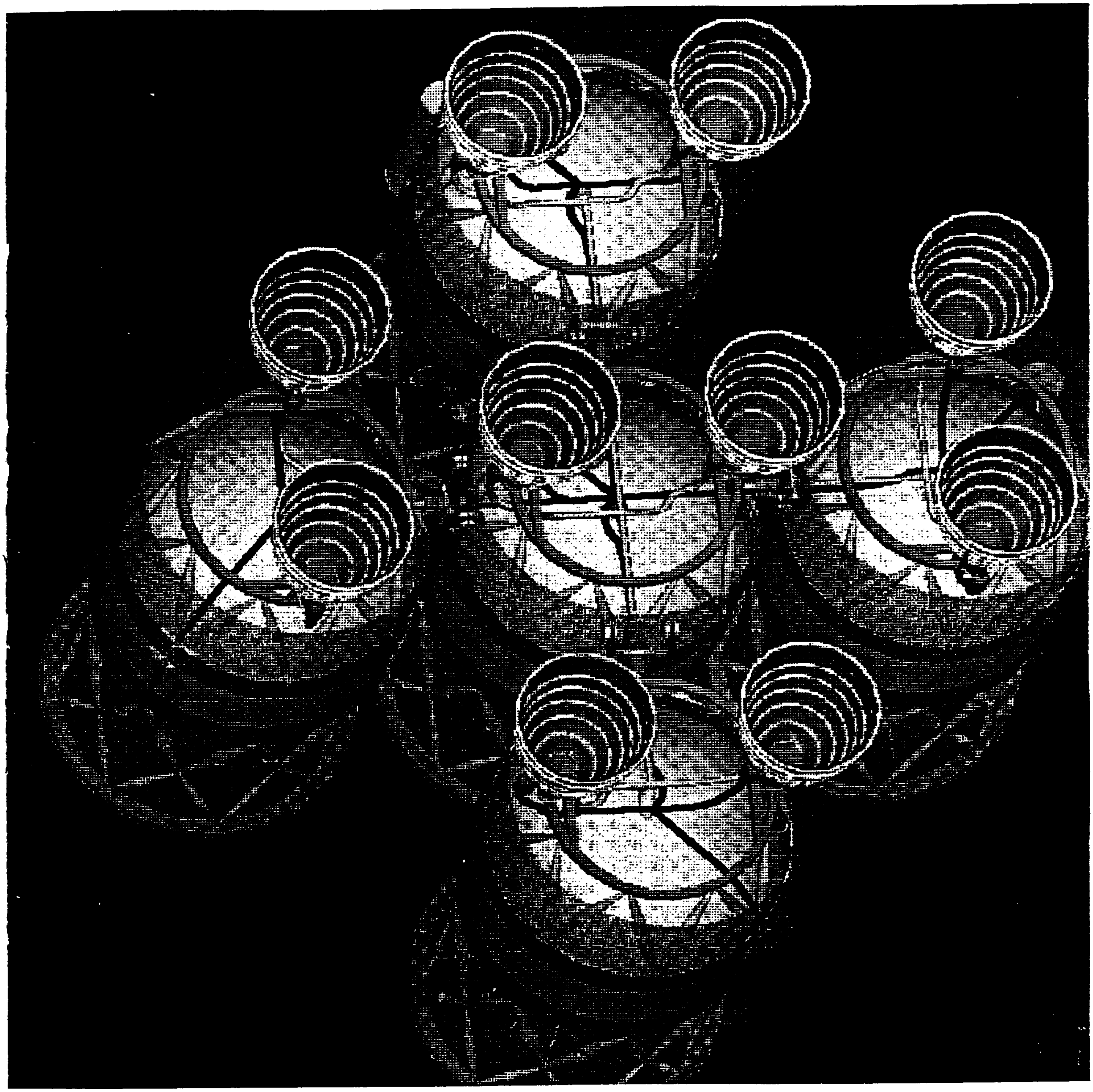

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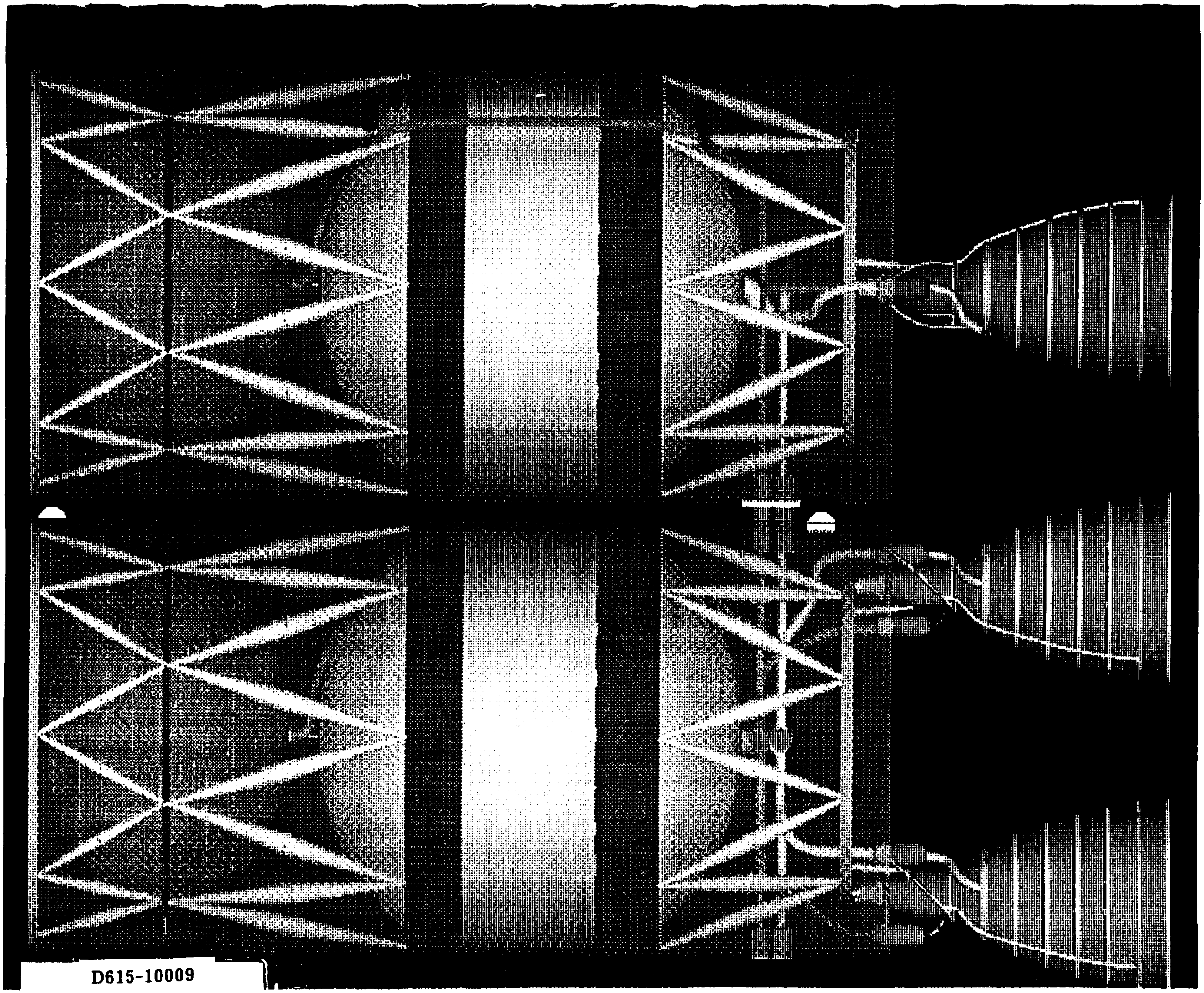

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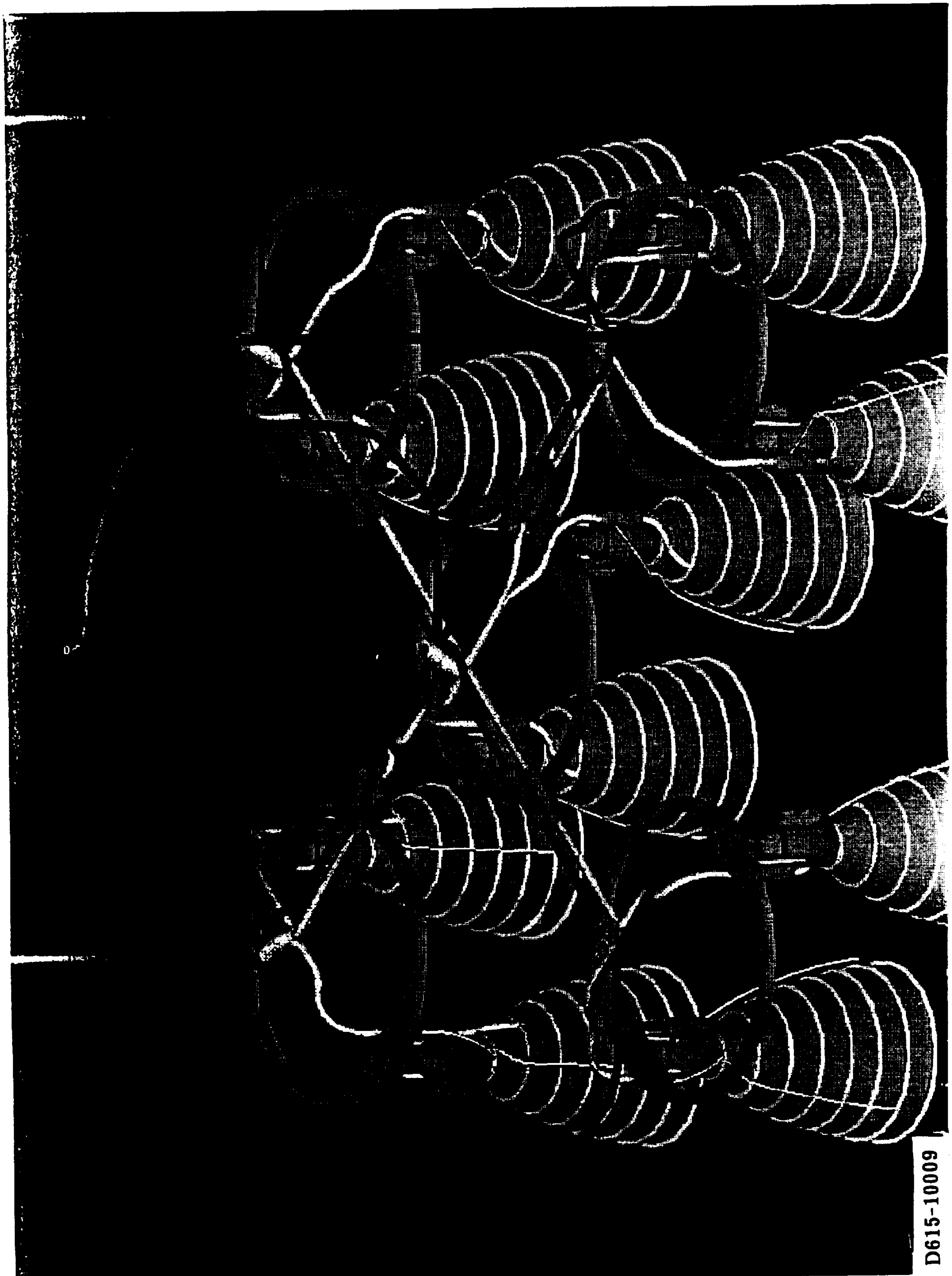
Dual-Engine Manifolded Plumbing

The maximum capacity configuration includes four strap-ons around the core unit, with a total of 10 engines and corresponding plumbing.



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C-3

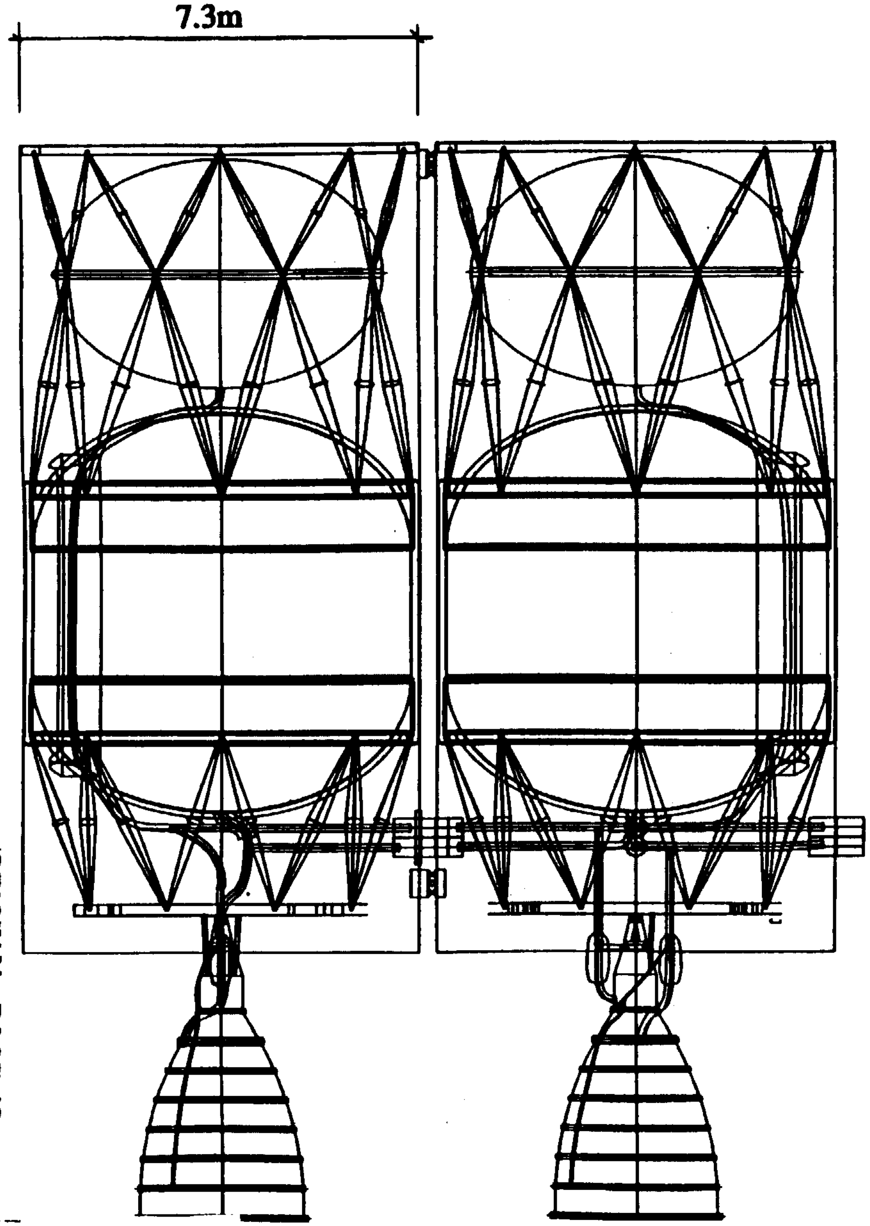
Shuttle-Z 3rd Stage TMIS
Single-engine Version

The concept was also investigated using a single-engine approach, to reduce parts count and plumbing complexity since 5 engines provide adequate thrust-to-weight on TMI. Disadvantages of this approach are no engine-out for each unit's own ETO ascent, and the additional requirement for a vernier roll-control system (which would consume the weight saving per unit available from using only one main engine).

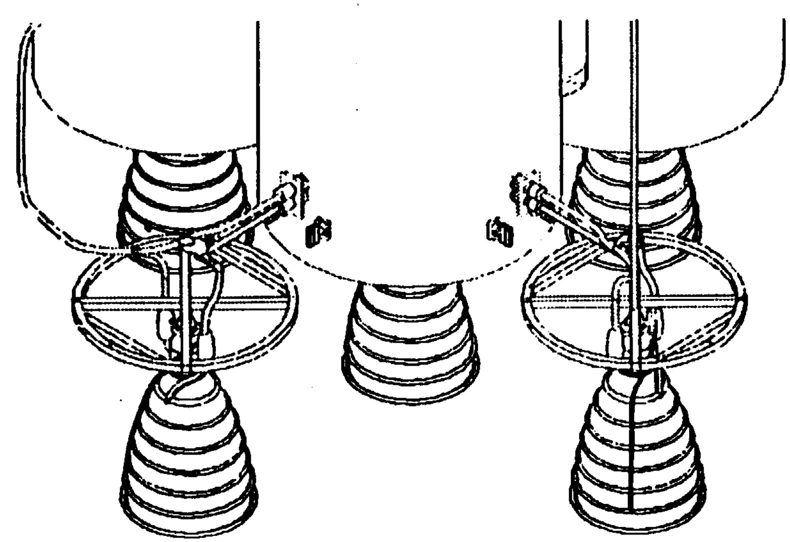
The following three charts show a 3-D model of this version, and are followed by the reference mass statement for the unit.

The Shuttle-Z 3rd stage TMIS concept cannot easily accommodate engine-out geometry for TMI in either version, because of the wide separation of its engines. When considered together with the reference cryo/aerobraked Mars mission vehicle components (MTV and MEV), either a 27° engine gimbal angle is required to track the composite mass center during all phases of the TMI burn, or a separation of 120 m between the TMIS and the MTV/MEV assembly. This latter approach is more feasible, as the required truss column would be relatively light and the maximum attainable gimbal angle is about 12°; however, such a truss would need to be accounted for in the mission mass statement.

"Shuttle-Z 3rd Stage" TMIS



- Fewer parts, less plumbing.
- No engine-out on each unit's orbital ascent.
- Modular TMIS accommodates 100-508 t propellant.
- "Fly-together" automated rendezvous & docking.
- Core unit: - Umbilical connections for up to 4 strap-on units.
  - Contains interconnect manifolds.
- Strap-on units: - All identical.
  - (1) 150 klbf engine.
  - 102 t propellant maximum.
- Vernier roll-control system on each unit.



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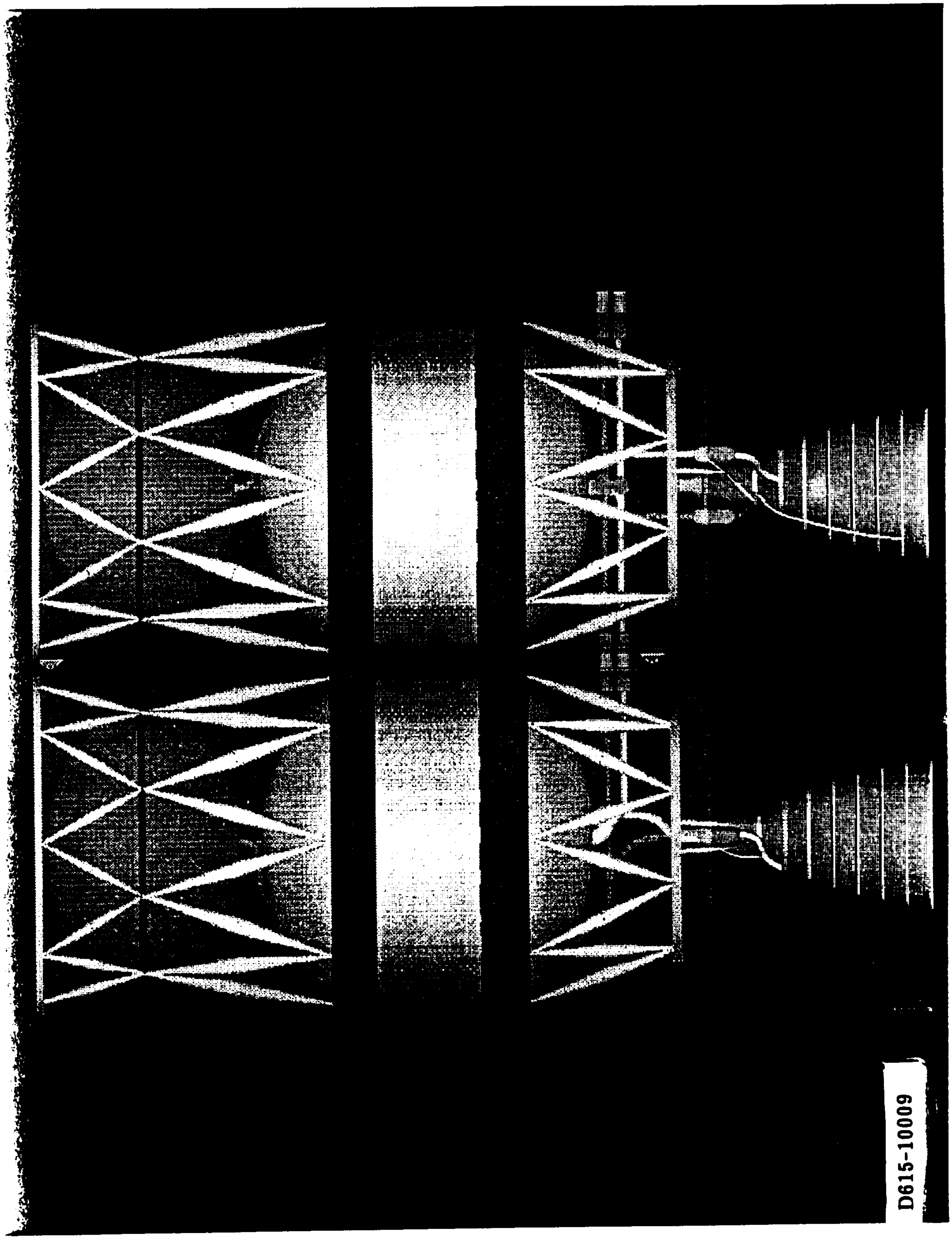
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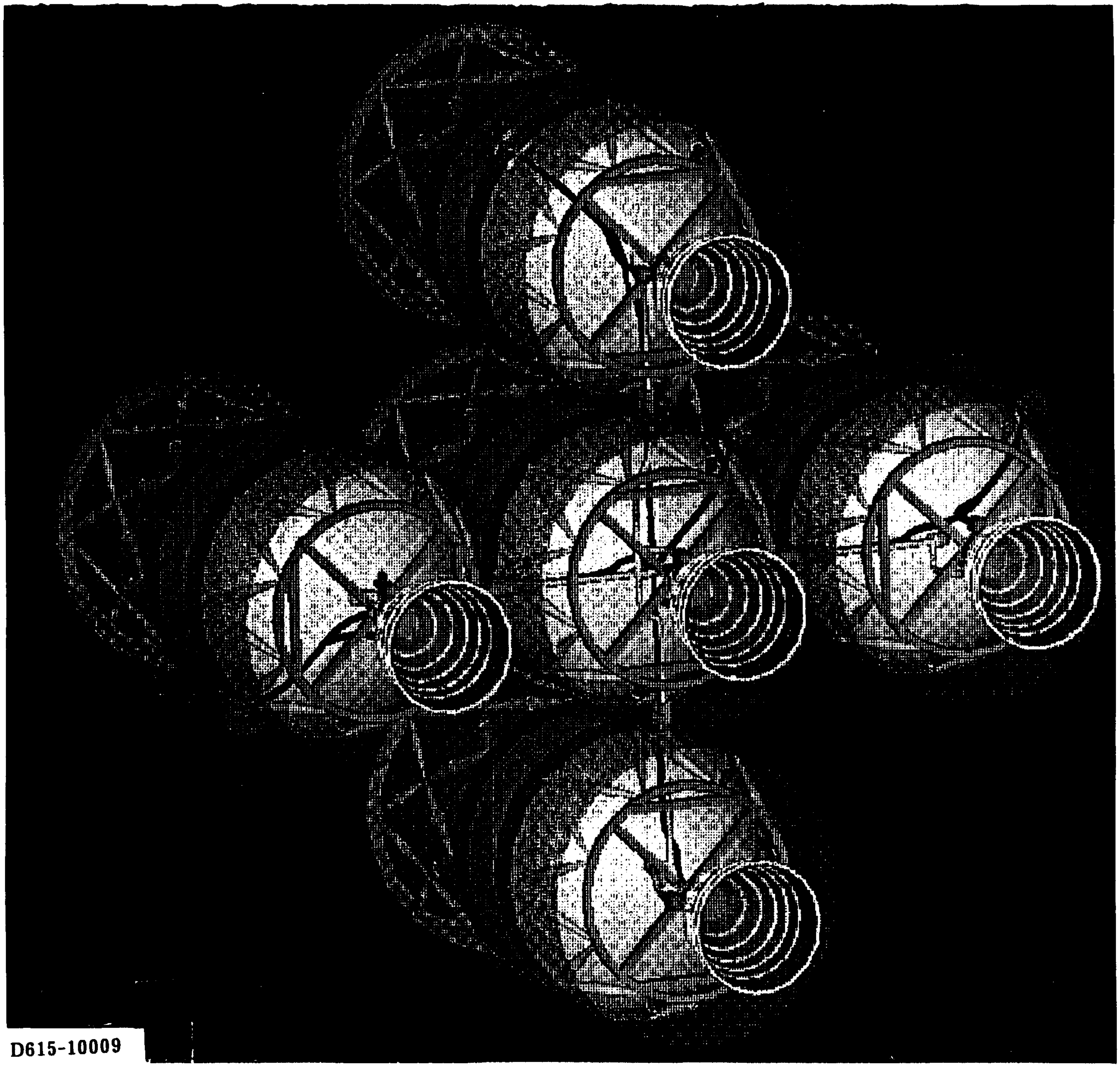

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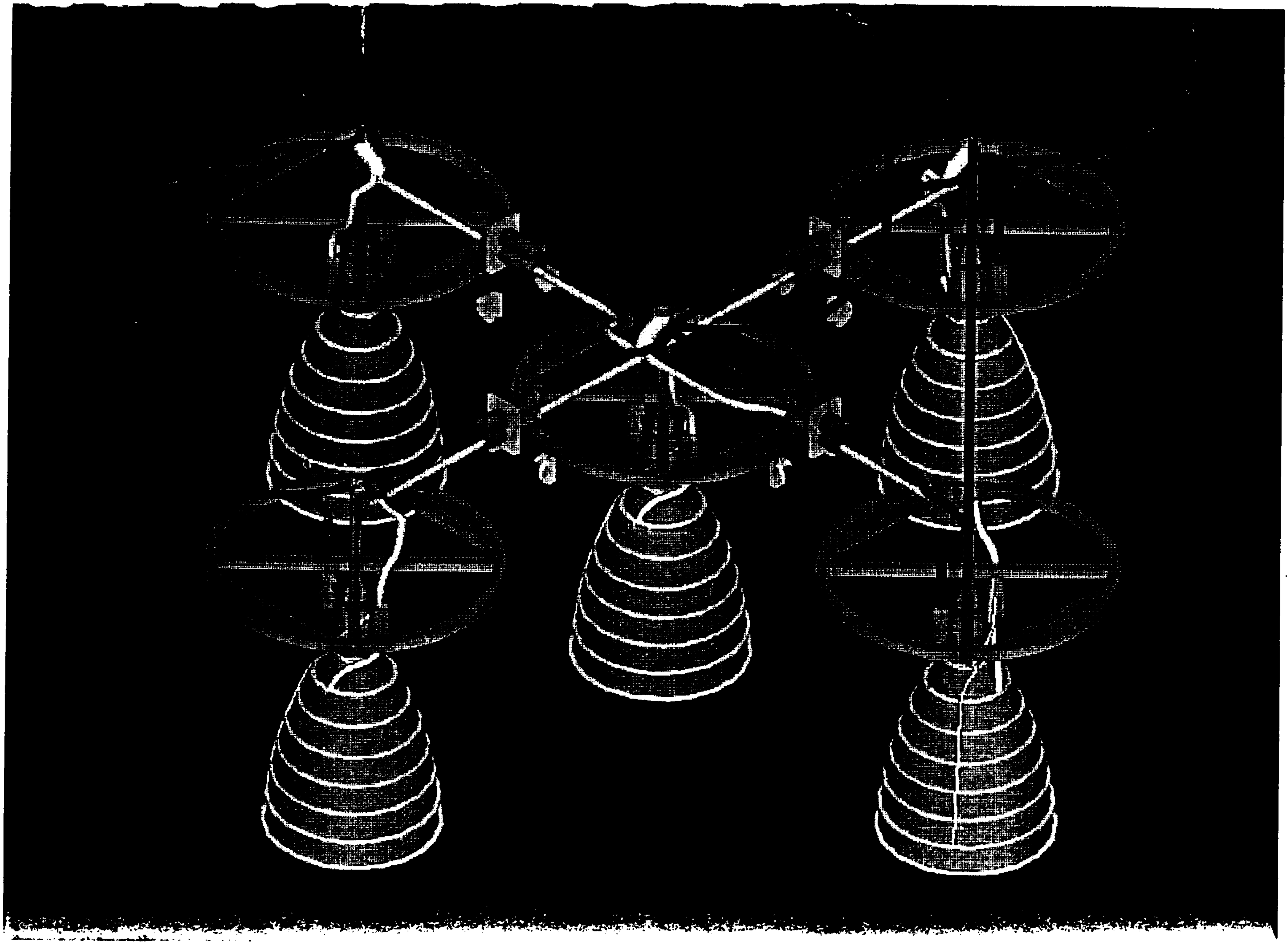

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ADVANCED CIVIL SPACE SYSTEMS

Shuttle-Z TMIS Mass Statement

BOEING
STCAEM/sdc/28Mar90

	Mass (kg)
Structures	
Thrust Ring	452
Thrust Struts	210
Intertank Struts	360
Forward Struts	140
Forward Load Ring	325
Shell/Meteroid Shield	1,426
Total	**2,913**
Propulsion	
LO2 tank	810
LH2 tank	1,438
Engines	1,800
Engine Installed	360
Main Feed Lines	103
Prevalves	20
Tank Vent Valves	20
Pressurization	30
Interconnect/Fill & drain	40
Total	**4,621**
Thermal	
MLI/VCS System	2,538
Base Heat	98
Total	**2,636**
Avionics	
C & DH	380
GN & C	150
Antennae	10
Total	**540**
Power	
Batteries	200
Distribution & Control	200
Total	**400**
Total Dry	11,110
Growth (10%)	1,111
TOTAL DRY MASS (kg)	**12,221**

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Long-Duration Habitat Trade Study

Brent Sherwood

Agenda

Introduction
Options
Geometry Analyses
Configuration Analysis
Mass Analyses
Other Factors
Conclusions

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Long-duration Habitat Trade Study
Contents (1)

Introduction

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Long-duration Habitat Trade Study
Contents (2)

Geometry Analysis

Topology Analysis Metrics (1 - 2)
Module Cluster Topology Analysis (1 - 5)
Topology Metric Analysis: Aerobrake Integration Factor
- Safe-Haven Split Factor
- Spatial Units Factor
- Parts Count Factor
- Proximity Convenience Factor
- Circulation Efficiency Factor

Long-duration Hab "Tunnel" Arrangements: 7.6 m-diameter Cross Section Properties
- 10 m-diameter Cross Section Properties
- 4.4 m-diameter Cross Section Properties

Geometry Analysis Metrics (1 - 3)
Habitation Module Geometry Metrics (1 - 2)
Geometry Metric Analysis: Inhabitability Factor
- Vault Factor
- Domain Factor
- Hallway Factor
- Spaciousness Factor
- Variety Factor
- Elevator Factor
- Perimeter Factor
- Pathway Factor
- Scale Factor

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Long-duration Habitat Trade Study Contents (3)

Long-duration Habitat Trade Study Summary

This chart summarizes the process and results of an extensive trade study to compare alternative concepts for long-duration habitats.

Long-duration Habitat Trade Study Summary

ADVANCED CIVIL SPACE SYSTEMS

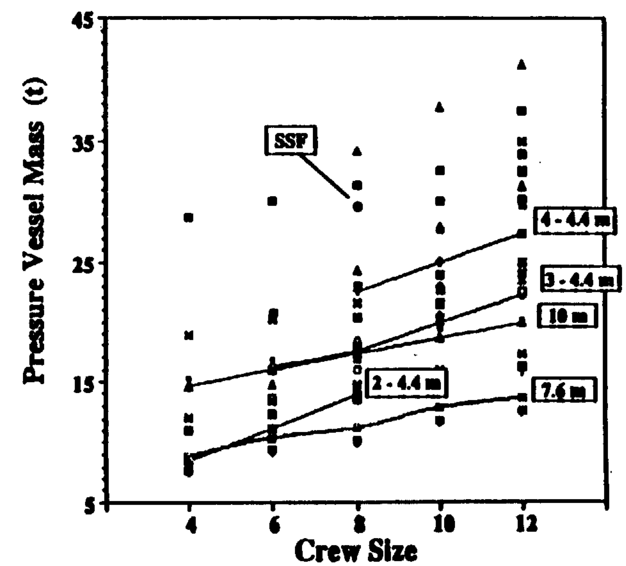
BOEING
STCAEM/bs/14Mar90

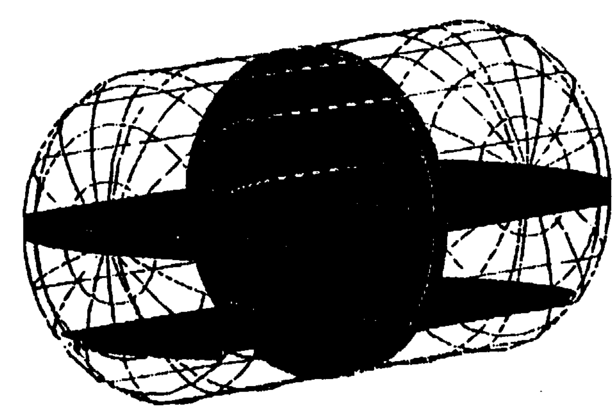
Process

- Trade space matrixed 5 crew sizes and 3 module sizes
- Generated 1480 distinct options, based on gravity, orientation, topology and structure; focused on 150 concepts
- Developed metrics for selecting preferred topologies and geometries; reference configurations for crew response survey
- Weighed pressure vessel structures, estimated equipment outfitted weights; assessed integration impact, commonality, growth potential, manufacturing options

Results

- Generated data allow applying a wide variety of priority sets to determine "optimal" concepts for specific architectures
- First HEI decade can use lightened SSF derivatives for all crew systems: LTV, LEV, surface outposts, safe-havens
- Later, long-duration missions require a larger module, common across architectures and capable of integration with smaller modules
- Trade neckdown led to synthesizing novel module concept, using best features from the studied options
- A 7.6 m diameter vessel, "tunnel-oriented", sized for 6 crew, with a cross-sectional bulkhead, was selected as the reference modular unit





Motivation

This chart explains how the long-duration hab trade study came about, and why its results are critical for further vehicle concept definition in the STCAEM study.

MTV Habitat Trade Study
Motivation

Why trade hab concepts? Mars Transfer Duration & Environment

- 1020 d design duration (SSF is 90 d)
- Deep space (SSF is in LEO)
- No escape, no resupply, no crew rotation

What are the major options? Module Size, Diameter & Number

- Space habitats have traditionally taken advantage of the maximum "throw" diameter of their launch vehicles -- HEI launch vehicles are large
- Volume is at a premium due to mass & packaging

Why is a choice necessary for STCAEM? Vehicle Integration

- Mass more critical for transportation systems than for LEO facilities
- Crew system is the MTV payload; comprises about 1/4 of MTV mass
- Sizes propulsion system, structure, aerobrake (if one)
- Constrains integrated vehicle configuration for some propulsion options

Habitat Module Evolutionary Context

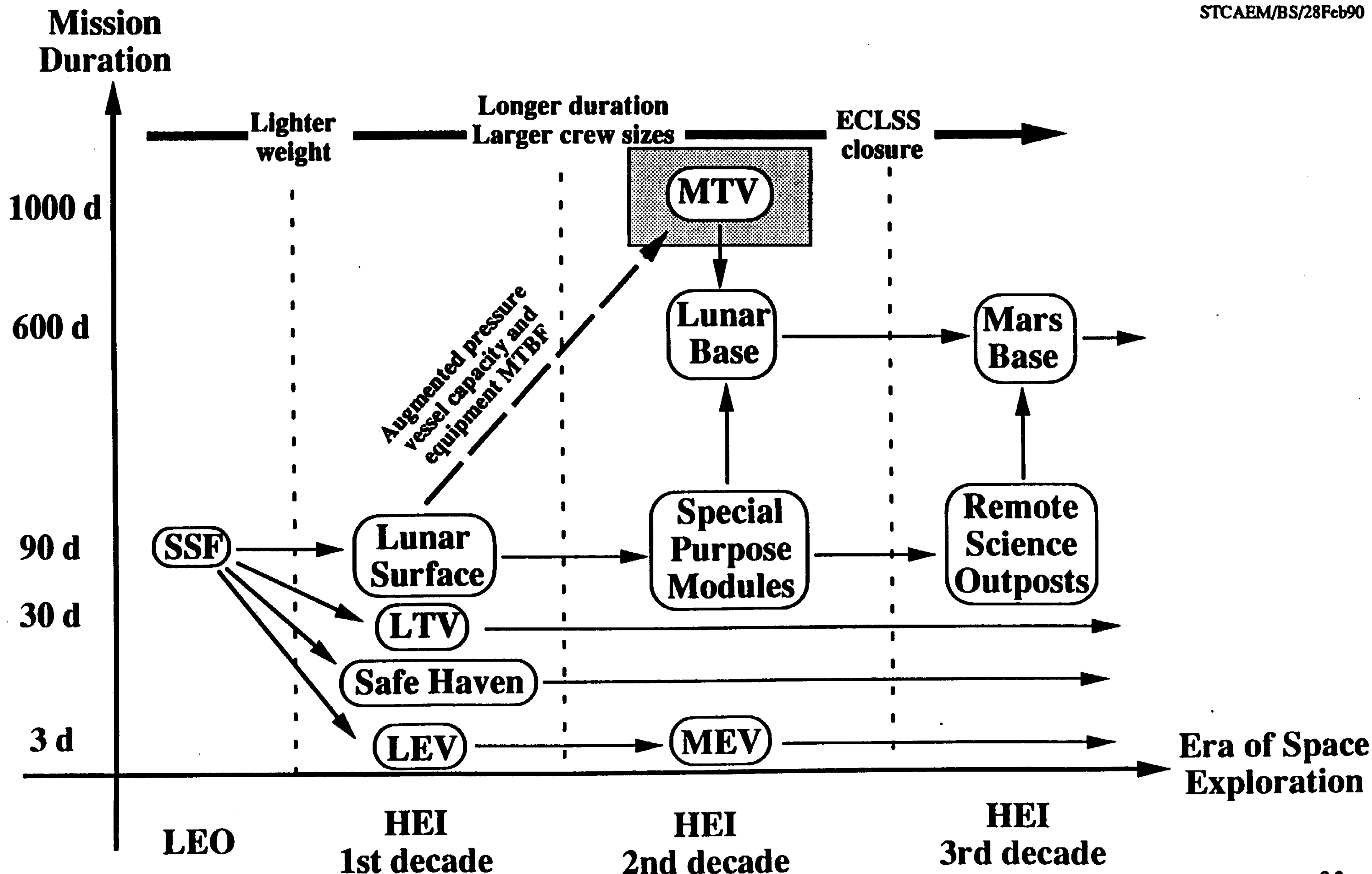
The gross division of HEI into three functional decades subsequent to the 1990s helps organize thinking about habitat system requirements. Here individual crew member mission duration is plotted against program phase, to generate a space populated by various kinds of habitation systems.

SSF-derivatives can serve a great many HEI functions, including crew cabs for LTV, LEV and MEV concepts as well as several kinds of unique applications in space and on planetary surfaces. The key new requirement is decreased structure mass, since uses depending on deep-space transportation are more sensitive to mass than are permanent LEO facilities like SSF.

Some applications, including consolidation-phase surface bases and especially the MTV, must be designed for crew-rotation durations an order of magnitude longer than those best served by the SSF-derivatives. Key new requirements are enhanced equipment reliability and augmented pressure vessel capacity. This trade study therefore concentrates on the MTV application, targeting extremely long durations and the 2nd decade of HEI operations.

Regardless of specific results, we would expect advanced habitation systems (such as planetary bases) to be comprised of both kinds (SSF-derived and advanced) of elements.

Habitat Module Evolutionary Context



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MTV Habitat Trade Study
Goals

- **Generate and evaluate a reasonably inclusive set of habitat options suitable for evolutionary, deep-space, long-duration missions**

- **Develop and apply a set of metrics which include criteria of:**
  Functionality (Does the concept work?)
  Perception (How will the crew respond?)
  Integration (Does the concept fit into the mission architecture?)
  Cost (Is the concept technologically and programmatically affordable?)

- **Develop and present the trade data transparently, so that they can be used for a variety of concept selections under different circumstances**

- **Determine criteria weighting appropriate for STCAEM goals: light weight, vehicle integration, evolution & growth, commonality**

- **Select a reference concept for immediate application in current MTV concept definition for the STCAEM Study**

MTV Habitat Trade Space

The fundamental trade space addressed by the study is displayed in matrix form here, plotting five crew sizes against three fundamental sizes of module. Because of the critical constraint of launch vehicle capacity, the candidate module diameters were chosen as: identical with SSF; the 25' diameter commonly discussed for an HEI Shuttle-C or a small ALS shroud, and the 33' diameter which has been suggested for a larger ALS shroud. The study spanned the trade space as shown with combinations of geometrical, weight and configuration analyses.

MTV Habitat Trade Space

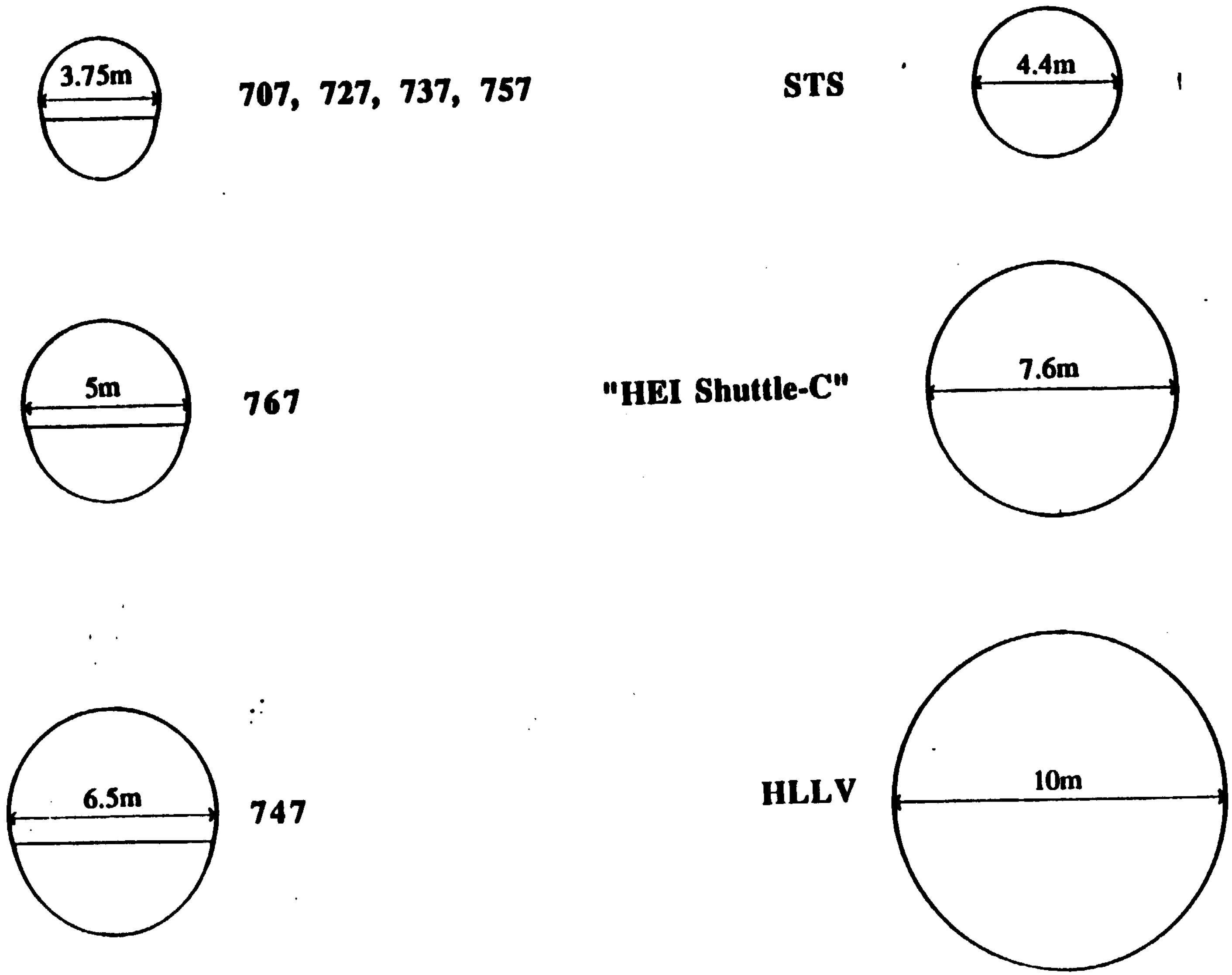
Module Type / Crew Size	Baseline SSF All equipment identical; "sending SSF to Mars"	4.4 m Diameter SSF diameter; saddle hatches; unpenetrated end domes; optimized rack frames;	7.6 m Diameter Lunar Shuttle-C diameter; other features unchanged	10 m Diameter Mars HLLV diameter; other features unchanged
4			90-d Study MTV hab concept	
6				
8	"Off-the-shelf" SSF configurations and weights			
10				
12				

Analysis techniques key: Structure weight, module geometry, topology | Weight, geometry & configuration

Pressurized Cabin Diameter Comparison

This diagram compares, to scale, the cross-sections of several familiar aircraft, the SSF-diameter module size launchable with the NSTS, and the larger-diameter options considered by this trade study.

Pressurized Cabin Diameter Comparison

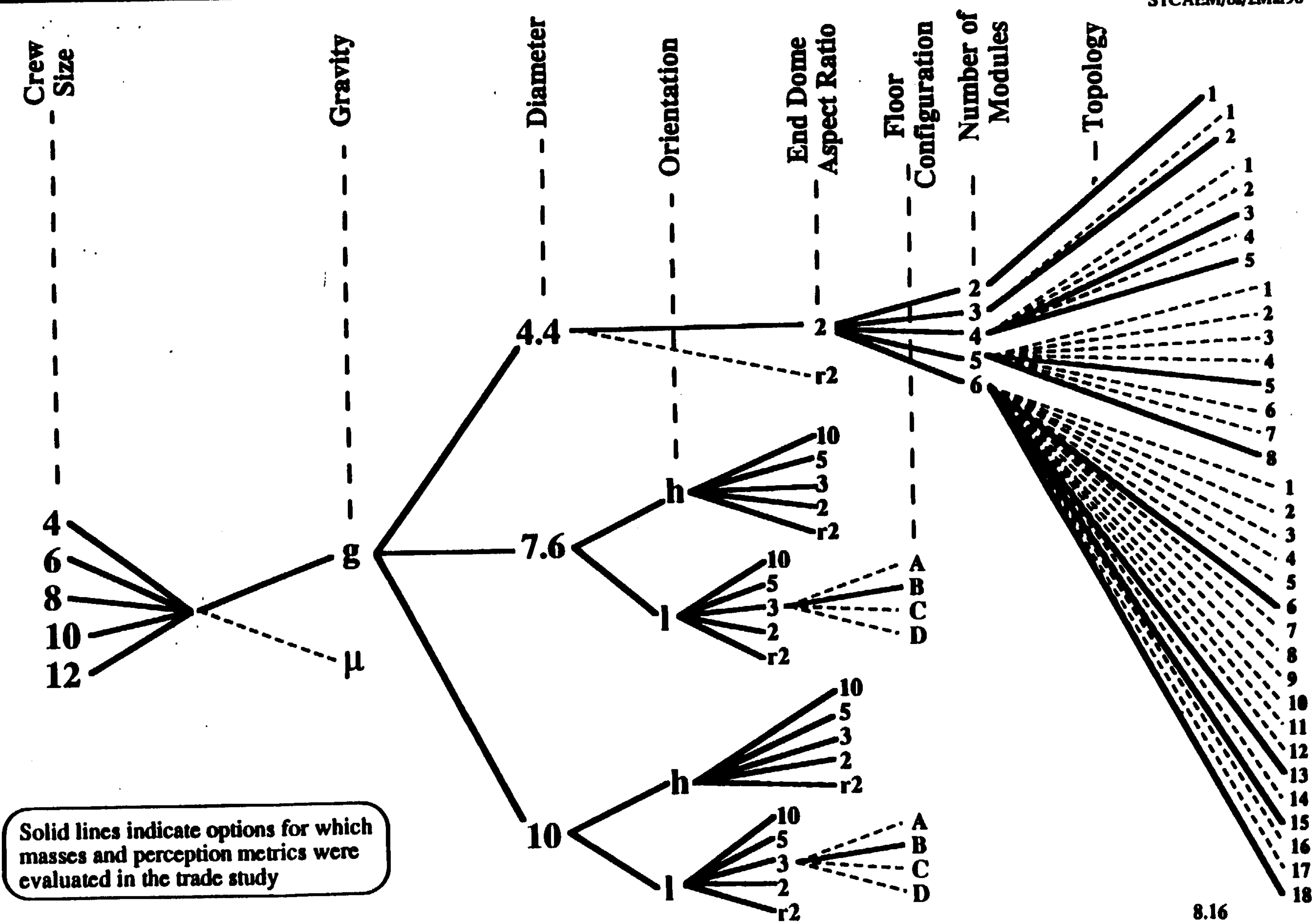


MTV Hab Trade Tree

This diagram shows the parameters varied in the study, to elaborate the trade space:
Crew Size
Gravity Requirement (binary alternative)
Diameter
Orientation of Floors (for the medium and large diameters only --- "h" means high, or stacked like sliced bologna; "l" means long, or arranged like a tunnel on its side)
End Dome Aspect Ratio (five options for the medium and large diameter modules; just two of those for the small module, approximating SSF module end shapes)
Floor Configuration (where in the circular cross section the floors are located for the tunnel-arranged medium and large diameter modules)
Number of Modules (in the clusters of small modules)
Topology (geometrical arrangement, and interconnection, of the cluster options)

The total number of distinct options generated by this trade tree is 1480.

MTV Hab Trade Tree



Solid lines indicate options for which masses and perception metrics were evaluated in the trade study

Habitat Concept Nomenclature

This chart explains the nomenclature used throughout the study to designate options.

ADVANCED CIVIL SPACE SYSTEMS

Habitat Concept Nomenclature

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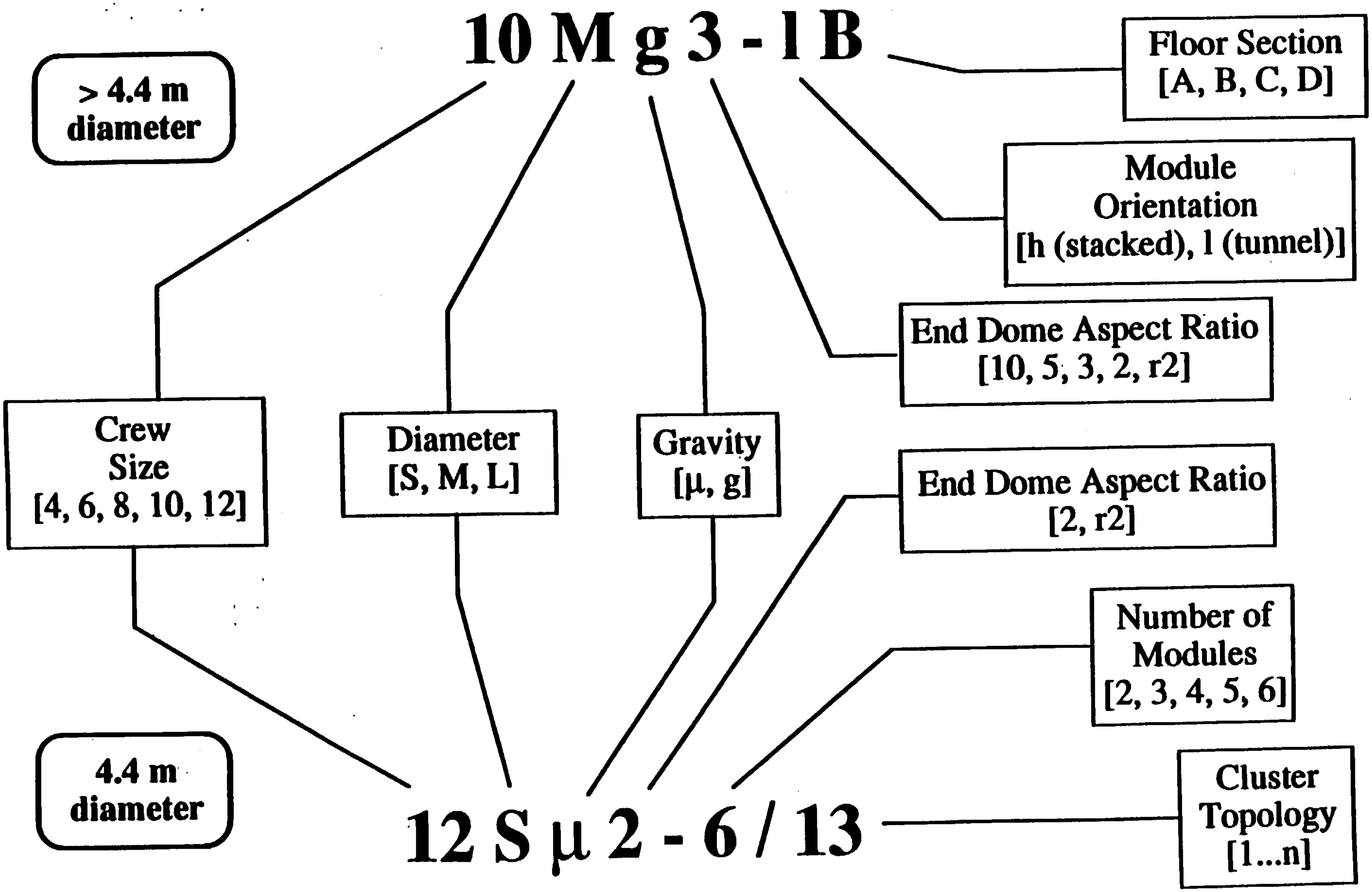

MTV Hab Trade Discriminators

Shown here, with non-exhaustive examples for clarification, are four categories of discriminators identified as dominant in the study.

MTV Hab Trade Discriminators

Functionality discriminators

- **Access** (proximities; maintenance; emergency)
- **Sensory interference**
  - Sound (variety; isolation)
  - Odor (galley; WMS)

Integration discriminators

- **MTV system implications** (aerobrake packaging; docking; assembly)
- **Growth potential** (evolution; larger transfer crews and vehicles)

Cost discriminators

- **Commonality** (SSF; planetary surface base use)
- **Manufacturability** (M&P; tooling)
- **Processing** (handling; outfitting)
- **Weight** (specific mass; total mass)

Perceptual discriminators

- **Proportion**
  - Volume (specific; total)
  - Articulation (shape; modulation; familiarity; versatility)
- **Scale**
  - Views (max sightlines; interior/exterior; choices)
  - Options (pathways; variety)

MTV Hab Trade Non-discriminators

Listed here, with exceptions, are the major characteristics and components identified as non-discriminators for the study. Specifically, effects of varying these "wash out" across the trade alternatives to first order, and so are not accounted for in the study.

MTV Hab Trade Non-discriminators

To first order, the effect of varying these components cancels across the habitat trade study

- **Internal configuration**
  (except as constrained by boundary condition)
- **Science payload equipment**
  (except access)
- **ECLS equipment selection**
  (except configuration)
- **Materials selection & finishes**
  (except M&P technology advances for primary structure)
- **Furnishings**
- **Hatches & windows**
  (used specifically for EVA)

MTV Habitat Trade Study Assumptions

Listed here are the governing assumptions made in the study to facilitate consistency in comparing the various options.

The issue of baselining gravity for long-duration spaceflight is largely sidestepped by this trade study (an artificial gravity impact assessment will be performed by STCAEM in the next quarter). We expect that the complications (mass and configuration) introduced by presuming artificial gravity constitute the superset, since artificial-g vehicles would still have μg flight regimes as well. Furthermore, the effort to exploit commonality between flight and surface habitation systems is best served by module designs which implicitly incorporate the presence of gravity. For these two reasons, quite independently of the possible physiological necessity or operational benefits of artificial gravity spaceflight, we emphasized the gravity options in this trade study. (The principal results which would be different are those which assume that, for instance, only one of the two end domes are available as "overhead" space. In μg, a single structure may serve as the "floor" for both spaces it divides.)

MTV Habitat Trade Study
Assumptions

- **Specific volumes according to extrapolated historical data** (excluding examples of aero-entry vehicles)
- **Space Station *Freedom* habitability standards as the point of departure** (SSF will provide the most sophisticated human environment to have flown in space)
- **"Stacked" module arrangement usable only for module diameters greater than SSF** (previous trades have shown inefficient utilization of space for vertical arrangements of small diameter modules)
- **2.3 m ceiling height used as standard** (for comparative purposes in the trade)
- **0.5 m floor thickness used as standard** (applied to medium and large diameter concepts, accommodates sound insulation & stowage)
- **All major hatch and window penetrations occur in barrel section** (minimizes mass, manufacturing complexity of end domes)
- **Cluster topologies contain no separate connecting nodes** (minimizes mass, vehicle packaging, parts count, additional procurement)
- **Module clusters use all same-length modules** (limits topology options to manageable number)
- **Galley / storm shelter structure integrated with floor structures above and below** (structural advantage of deep-beam configuration to keep weight down)
- **Gravity-condition options emphasized** (higher outfitted weight; must also accommodate μg regimes; result facilitates commonality with surface applications)

Volume Guidelines

These specific volume curves were assembled from historical sources, and are based on total pressurizable volume (without actual equipment solid volume subtracted). The STCAEM reference specific volumes for the MEV and MTV have been included.

Two features are notable. First, vehicles for which aeroentry was the dominant cabin configuration constraint have typically crowded their crews more than strictly in-space, habitation and non-capsule systems. Second, *Skylab* was not as anomalous as is traditionally claimed; *Freedom* has as much specific volume when hab, lab, all nodes, JEM and ESA modules are included. The key difference is that SSF has much more internal equipment than did *Skylab*, so the free volume is comparatively much smaller.

The upper curve can be used to choose specific volume for new module concepts, based on historical trends.

MTV Hab Trade Volume Guidelines

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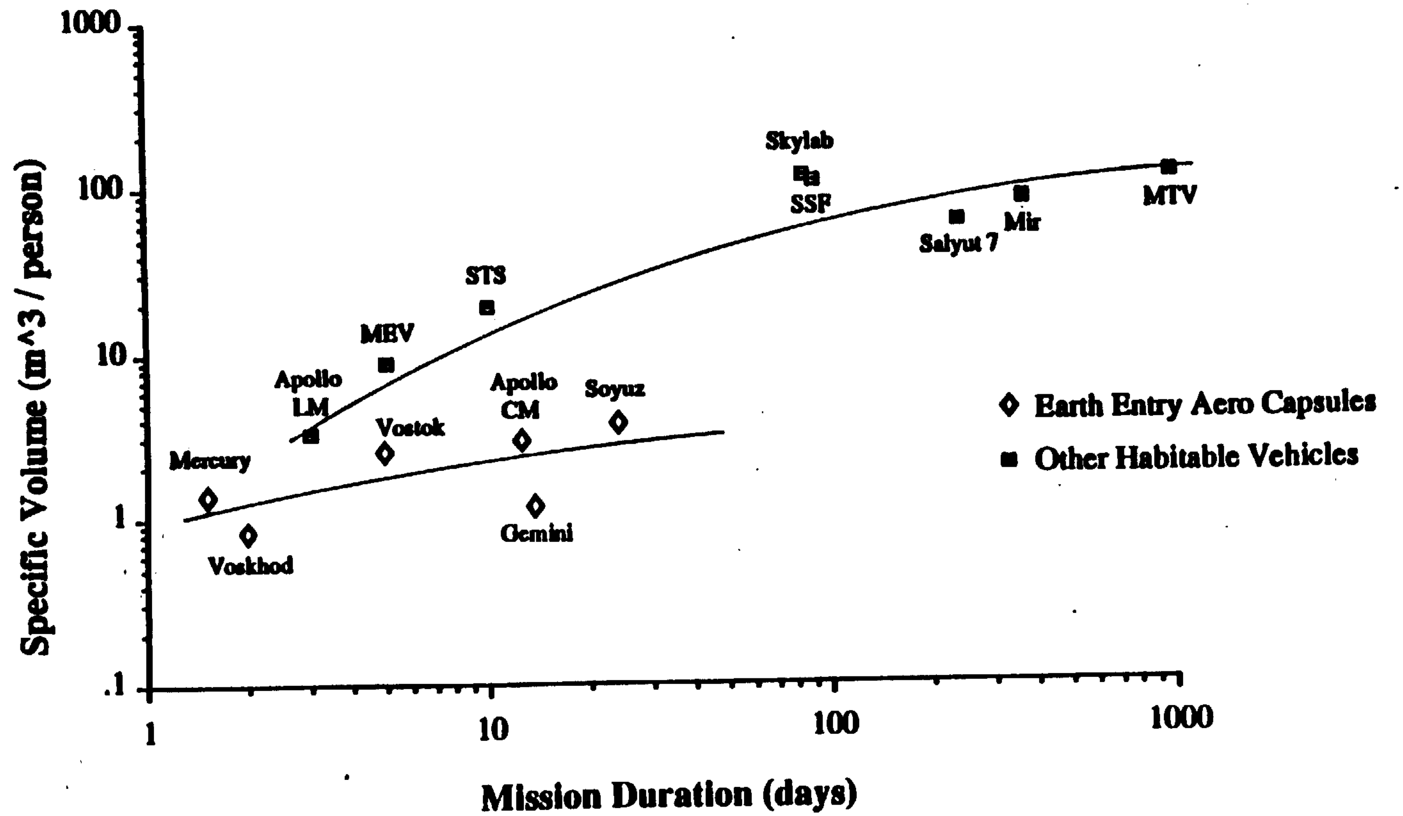
Historical Spacecraft Total Pressurized Volume Data



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Apollo News Reference, Grumman, 1989
Bluth, B.J. Soviet Space Stations as Analogs (NAGW-659), 1986
Boeing SSF WP-01 Data, 1990
Boeing STCAEM Study Data (NAS8-37857), 1990
NASA/JSC Man-Systems Division Data, 1989
Silvestri, G. et al. Quest for Space, 1985

Module Structure Concept Guidelines and Assumptions

Structural approaches are compared here, in several subsystem categories, for SSF and for this habitat trade study. Several subtle advancements have been introduced from the SSF approach, all in the service of reduced mass (which has extremely high leverage for in-space transportation systems). The heavy end cones are replaced by simple, unpenetrated ellipsoidal end domes (aspect ratio to be traded in this study); all module penetrations are in the less geometrically complex barrel section. The barrel sections are of monocoque, rather than waffle-grid, construction. The current precedent for this is in the (thicker-walled) bulkheads surrounding the dock port adapters of the SSF nodes. The advanced concept presumes overpressurization for structural stiffness on ETO launch, assuming ETO launch occurs using unmanned vehicles (e.g. Shuttle-C). This weight-reduction technique is less compatible with manned launchers like the NSTS.

ADVANCED CIVIL SPACE SYSTEMS

Module Structure Concept Guidelines and Assumptions

Component	Space Station *Freedom*	Reference Trade Study Structure Concept	Comments
Material	2219 - T8 Al (as-welded)	Same	Long experience; ult. strength - 38 ksi
Cylinder	45° waffle grid	Monocoque	SSF uses man-rated, side-mounted launch configuration, not overpressurized for structural rigidity
Cylinder Cap	25° Conical, with flat pressure bulkhead	Ellipsoidal, with no penetrations	Docking loads, assemblage stiffness, axial penetrations drive SSF design
Support Structure	Longitudinal support beams for launch loads; cylinder support rings	cylinder support rings; Intermodule support structure (4.4 m dia.)	Overpressure provides structural integrity for reference unmanned ETO; intermod. support for uneven bending loads on hab system structure
Pressure Bulkhead	Monolithic, integrated into endcones	Al/Al honeycomb (10 & 7.6 m dia.); SSF derived (4.4 m dia.)	Monolithic bulkhead mass prohibitive for large diameter; honeycomb lighter, with acceptable volume penalty
Module Connection	Pressurized nodes	Parallel tunnels with pressure bulkheads between modules (4.4m dia)	Mass critical for reference; no req't for growth flexibility of individual system

Note: Reference habitat structure design guidelines derived from MSFC-HDBK-505 Rev. A. Structural Strength Program Requirements.

Representative Geometry Options to Scale

The next five charts show, one for each of our crew sizes, comparisons to scale of the S cluster options and the M-1 and L-1 options.

ADVANCED CIVIL SPACE SYSTEMS

Representative Geometry Options to Scale
4 Crew

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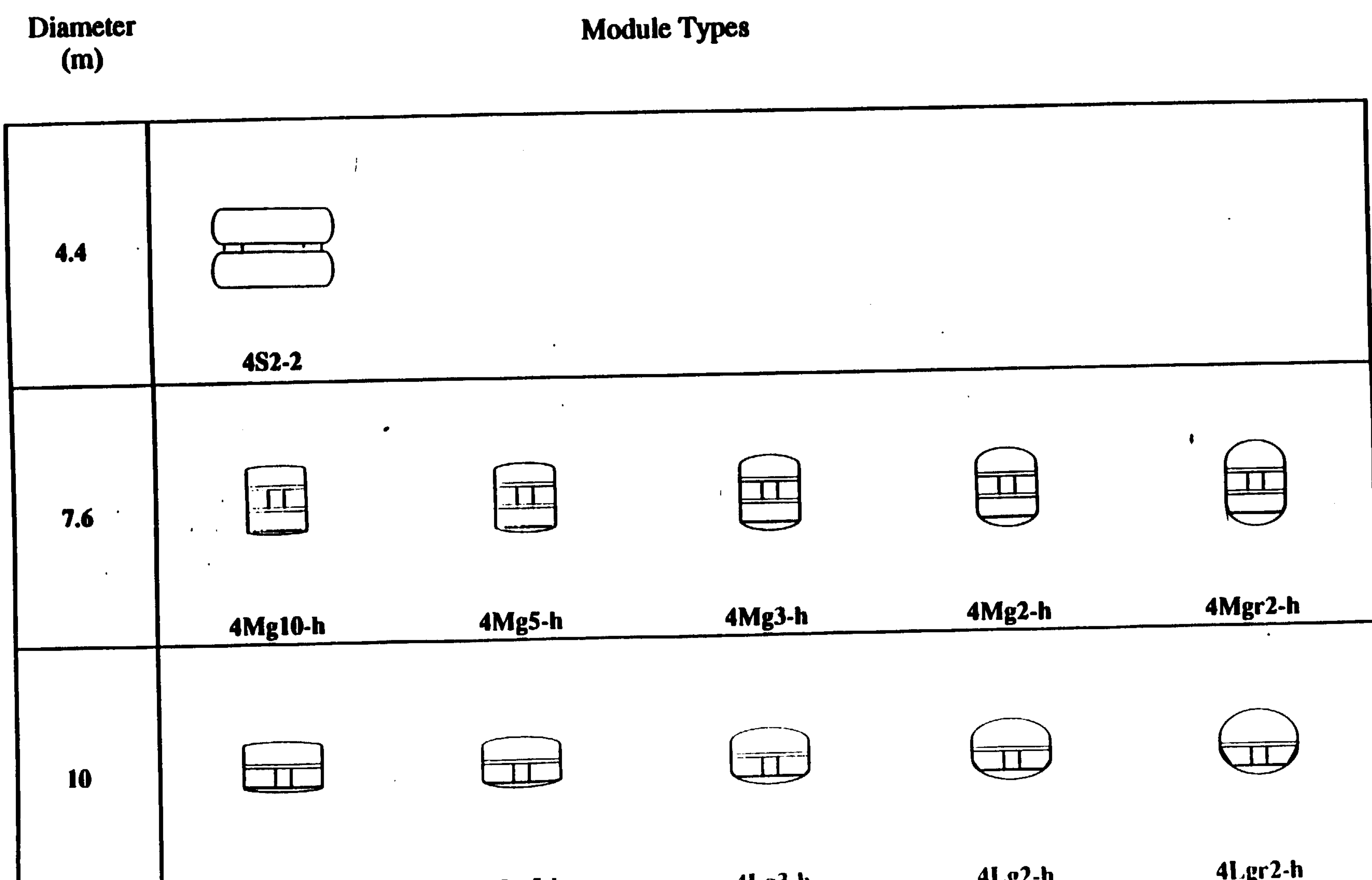

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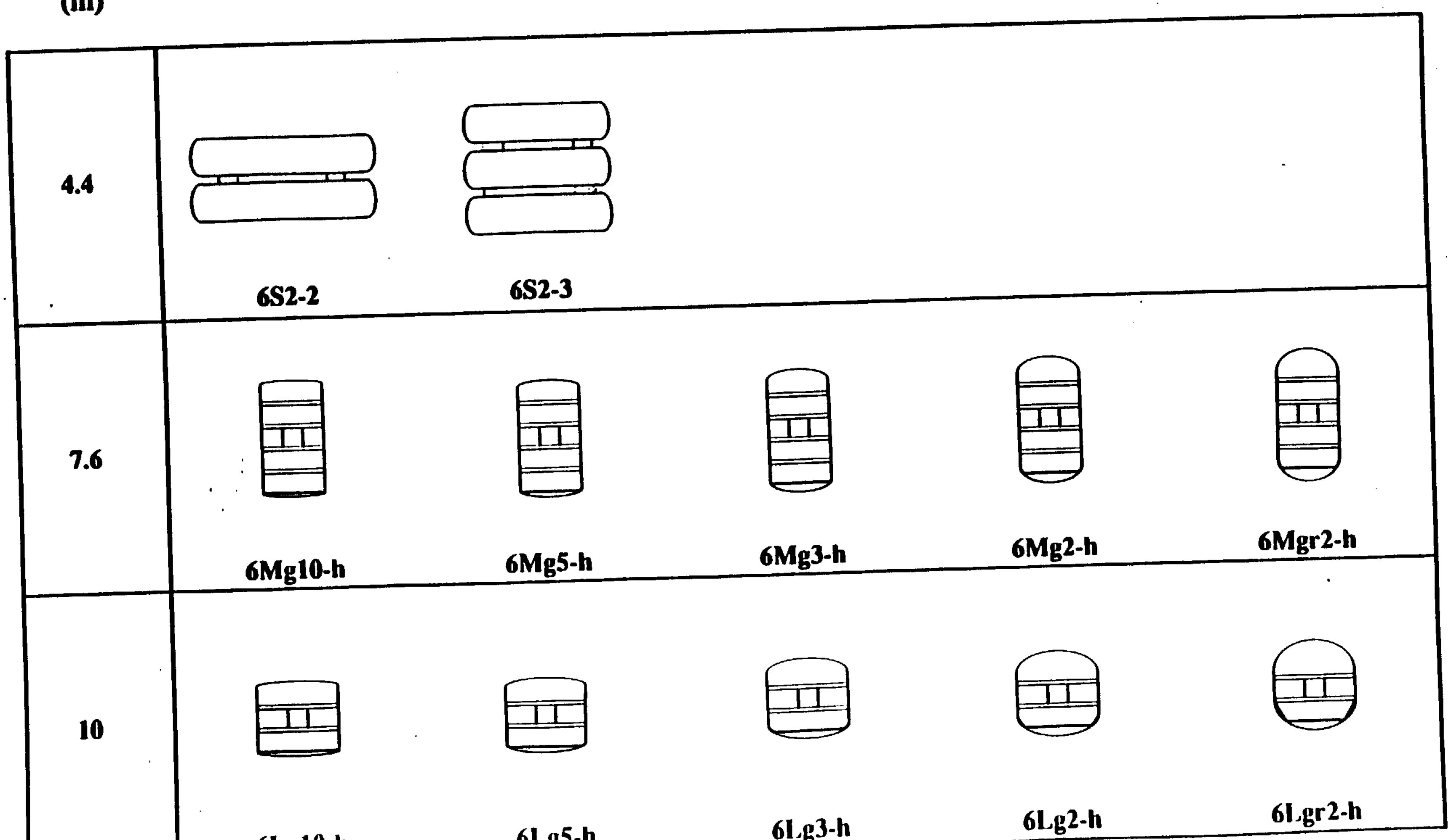
Representative Geometry Options to Scale
6 Crew

Diameter (m)

Module Types



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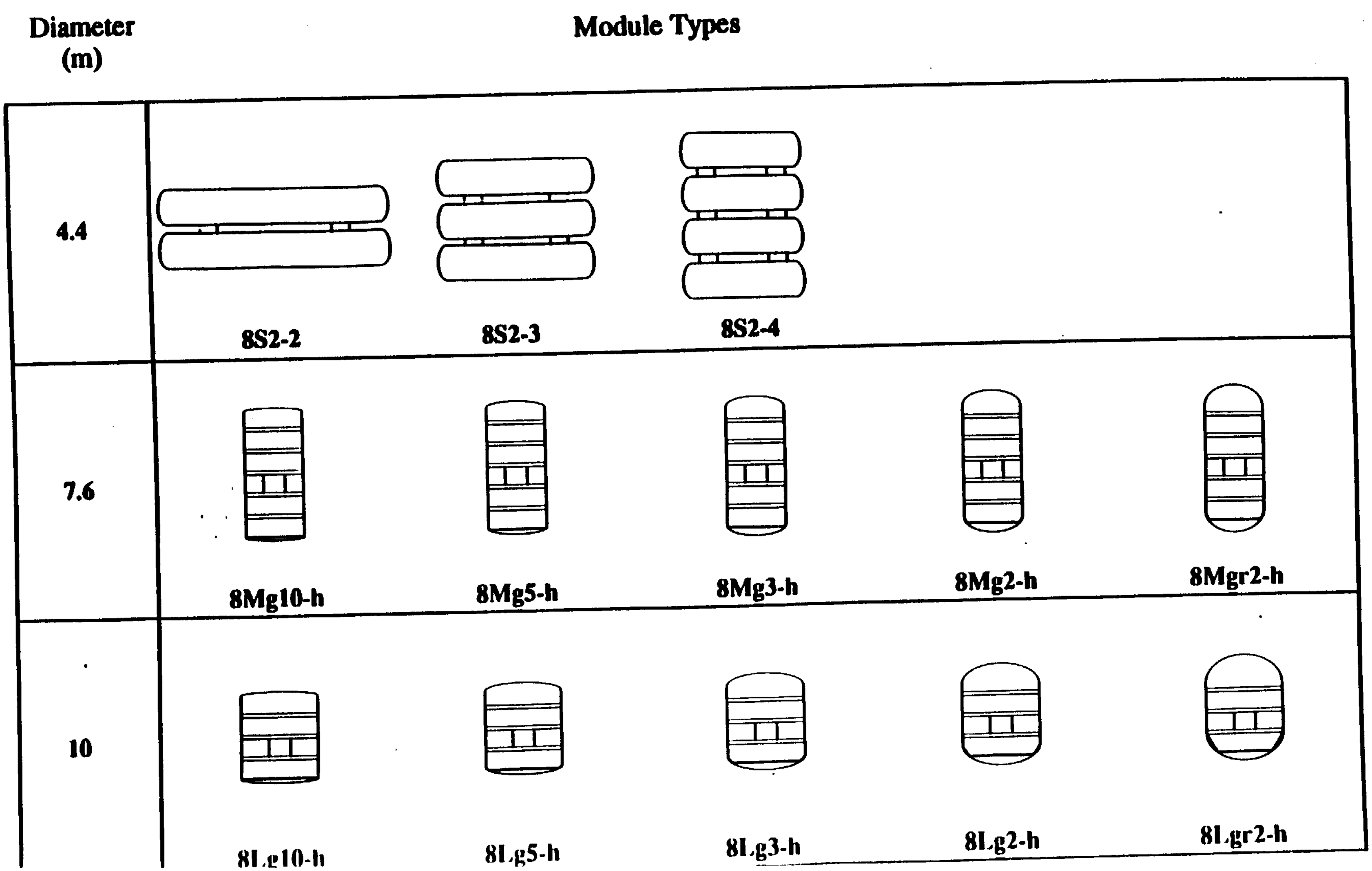
Representative Geometry Options to Scale
8 Crew



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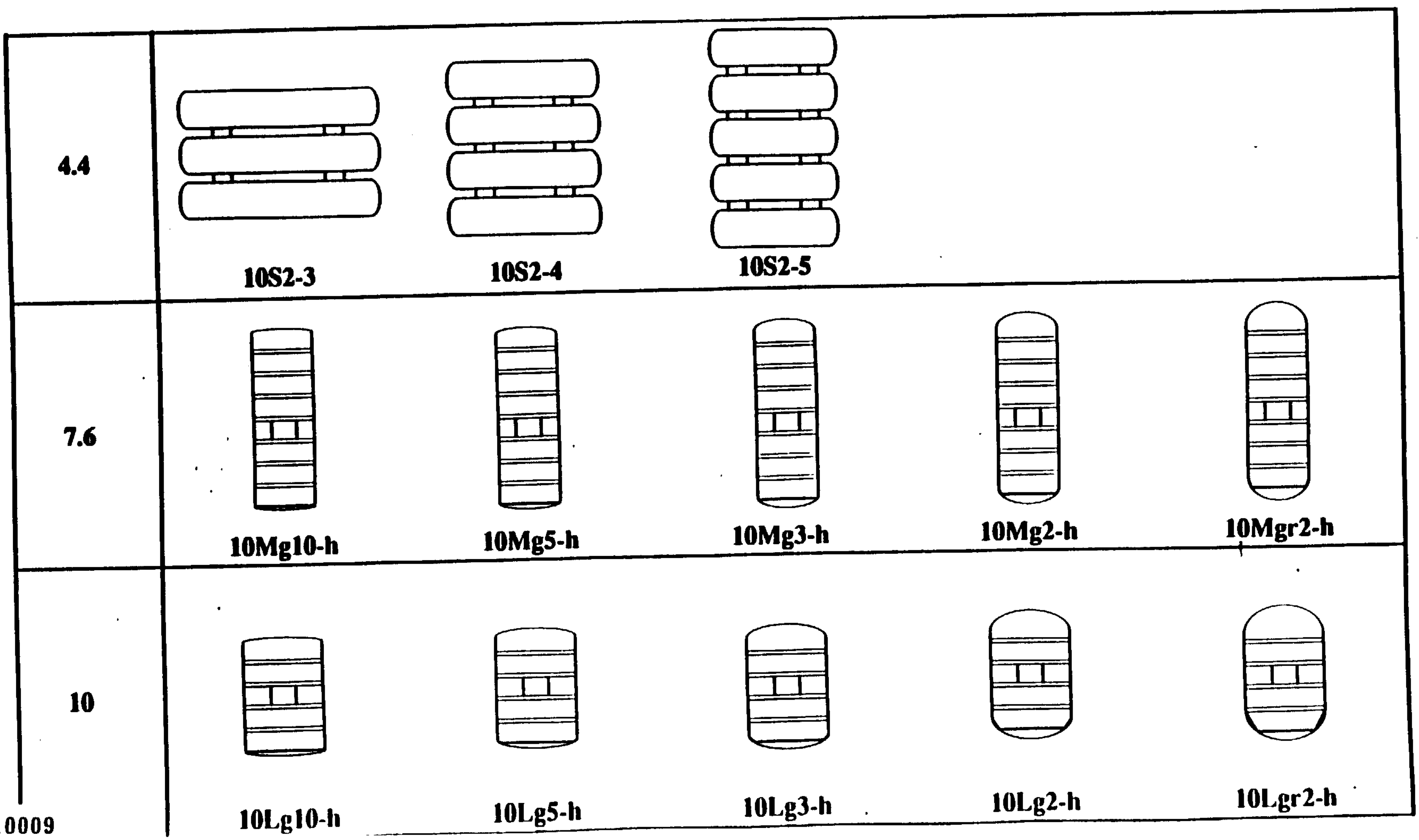
Representative Geometry Options to Scale
10 Crew

Diameter (m)

Module Types



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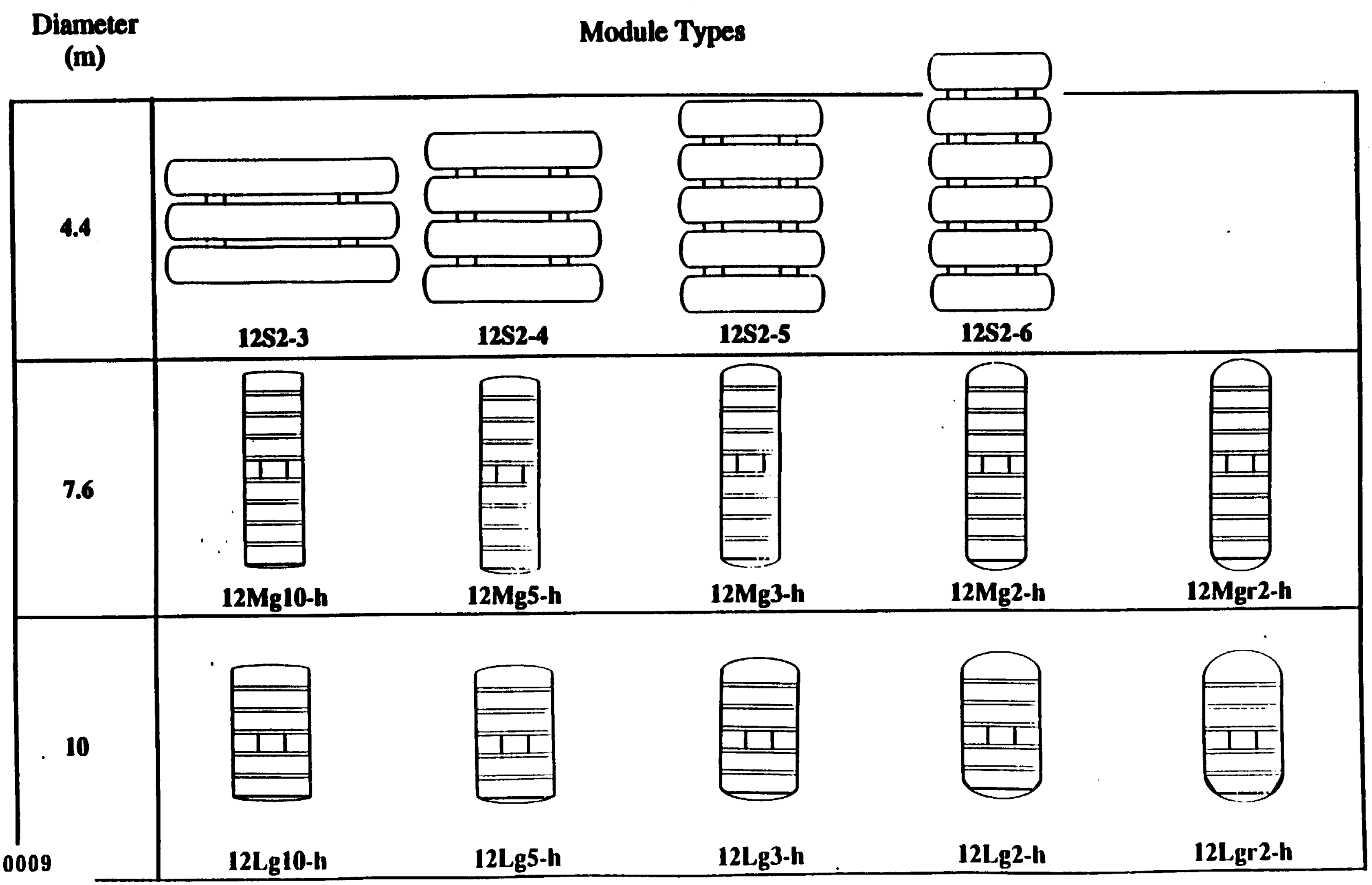
Representative Geometry Options to Scale
12 Crew



Topology Analysis Metrics

The first of the geometrical analyses is a topology study for cluster configurations of small modules. Seven metrics devised to compare the many cluster topologies are defined on the next two charts. The ranking priorities assigned them are not fixed, but simply record value trends used to neck down the options in this study. The metrics were applied both sequentially and simultaneously to reduce the starting option set to a manageable subset for the purposes of mass analysis later. The most critical metrics for the neckdown were F_i and F_{ss}. The others were used primarily to investigate quantitatively some characteristics of the topologies.

Topology Analysis Metrics

Nomenclature

$n \equiv$ number of modules
n_{ss} = number of modules available under safe-haven conditions
$t \equiv$ number of connecting tunnels
$i \equiv$ each starting module for a circulation pattern, (1...n)
$j \equiv$ each destination module, (1...n-1)

Guidelines

- Each metric ranges between 0 and 1
- For each metric, higher is better

Symbol	Metric name	Definition	Comments	Ranking priority
F_{ew}	ECLSS weight	0 if 4 strings 1 if 3 strings	– 2 fault tolerant requirement favors racetrack topologies – Extra string weight relatively small	1
F_i	Aerobrake integration	Ranked based on configuration experience: 0.1 - 0.9	– Lower numbers are hard to package behind L/D = 0.5 aerobrakes while keeping brake size small – Changes with aerobrake L/D	2

(continued)

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Topology Analysis Metrics (2)

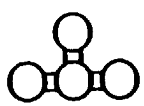
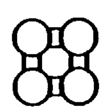
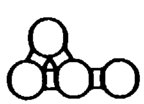
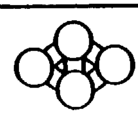
Symbol	Metric name	Definition	Comments	Priority
F_{ss}	Safe-haven split	$\frac{n_{ss}}{n}$	– Worst-case safe-haven condition for best possible ECLSS string distribution among n – Measures how much of the habitable volume is left (for remainder of trip if fix is impossible)	3
F_n	Normalized spatial units	$\frac{n}{6}$	– Normalized to n_{max} –Higher numbers mean greater potential for differently optimized environments	7
F_{pc}	Parts count	$\frac{1}{n+t}$	– higher numbers mean fewer pieces to integrate on-orbit, fewer mechanisms to maintain, less cabin air leakage	6
F_{pr}	Proximity convenience	$\left[\sum_{i=1}^{n}\sum_{j=1}^{n-1} t\right]^{-1}$	– Higher numbers mean fewer tunnels stand between origin and destination, summed over the topology – High numbers mean more convenience – Lower numbers mean potentially greater perception of inhabited domain	5
F_c	Circulation efficiency	$\frac{n}{t}$	– High numbers mean fewer connecting tunnels – Lower numbers may indicate "excessive" tunnels	4

Module Cluster Topology Analysis

The next five charts diagram the topologies considered for clusters of two, three, four, five and six modules. The topology metrics calculated are tabulated, and the topologies selected for further consideration (as representative of the best candidates from each group) are indicated.

The subsequent six charts graph the six most revealing metrics, to compare all the topology options.

Module Cluster Topology Analysis (1)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
2	2/1 	2	4	0	.9	0.5	.33	.25	0.500	1
3	3/1 	4	4	0	.7	0.33	.5	0.143	0.125	.75
	3/2 	6	4 3	0 1	.9	0.67	.5	0.111	0.167	.5
4	4/1 	6	4	0	.5	0.5	.67	0.100	.050	.67
	4/2 	6	4	0	.7	0.25	.67	0.100	.056	.67
	4/3 	8	4 3	0 1	.65	0.75	.67	0.083	.063 0.063	.5
	4/4 	8	4 3	0 1	.6	0.5	.67	.083	.063	.5
	4/5 	10	4 3	0 1	.8	0.75	.67	.071	.071	.4

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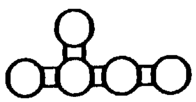
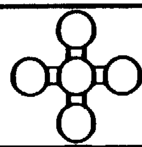
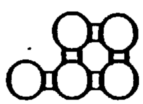
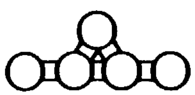
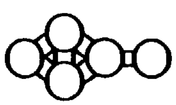
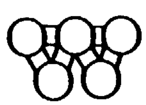
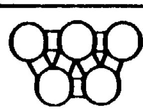
Module Cluster Topology Analysis (2)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
5	5/1 	8	4	0	.3	0.4	.83	.077	.025	.63
	5/2 	8	4	0	.5	0.4	.83	.077	.028	.63
	5/3 	8	4	0	.2	0.2	.83	.077	.031	.63
	5/4 	10	4 3	0 1	.5	0.6	.83	.067	.031	.5
	5/5 	10	4 3	0 1	.5	0.6	.83	.067	.031	.5
	5/6 	12	4 3	0 1	.5	0.6	.83	.059	0.33	.42
	5/7 	12	4 3	0 1	.7	0.4	.83	.059	0.36	.42
	5/8 	14	4 3	0 1	.7	0.6	.83	.053	0.38	.36

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Module Cluster Topology Analysis (3)

ADVANCED CIVIL SPACE SYSTEMS

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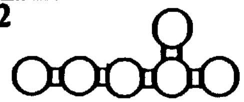
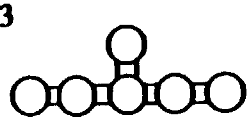
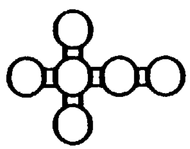
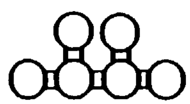
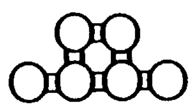
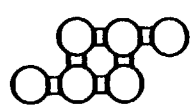
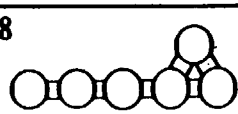
Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
6	6/1 	10	3	0	.1	0.5	1	.063	0.014	.6
	6/2 	10	4	0	.2	0.5	1	.063	0.016	.6
	6/3 	10	4	0	.3	0.33	1	.063	0.017	.6
	6/4 	10	4	0	.2	0.33	1	.063	0.018	.6
	6/5 	10	4	0	.5	0.5	1	.063	0.017	.6
	6/6 	12	4 3	0 1	.5	0.67	1	.056	0.019	.5
	6/7 	12	4 3	0 1	.3	0.67	1	.056	0.017	.5
	6/8 	12	4 3	0 1	.25	0.33	1	.056	0.016	.5

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ADVANCED CIVIL SPACE SYSTEMS

Module Cluster Topology Analysis (4)

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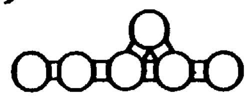
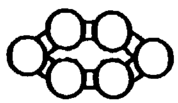
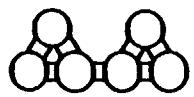
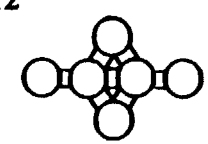
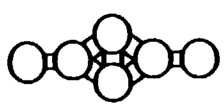
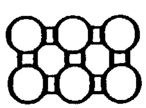
STCAEM/jeb/23Feb90

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{cw}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
6	6/9 	12	4 3	0 1	.3	0.5	1	.056	0.017	.5
	6/10 	12	4	0	.7	0.83	1	.056	0.019	.5
	6/11 	14	4 3	0 1	.5	0.5 0.33	1	.05	0.019	.43
	6/12 	14	4 3	0 1	.45	0.83 0.67	1	.05	0.021	.43
	6/13 	14	4 3	0 1	.45	0.83 0.67	1	.05	0.019	.43
	6/14 	14	4 3	0 1	.5	0.83	1	.05	.020	.43
	6/15 	16	4 3	0 1	.55	0.83	1	.045	.022	.38

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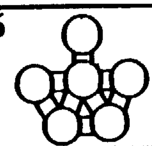
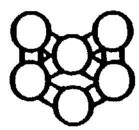
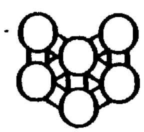
Module Cluster Topology Analysis (5)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
6	6/16 	16	4 3	0 1	.7	0.83	1	.045	.023	.38
	6/17 	16	4 3	0 1	.5	0.83	1	.045	.023	.38
	6/18 	18	4 3	0 1	.5	0.83	1	.042	.024	.33

• designates topologies included for further analysis

Topology Metric Analysis
Aerobrake Integration Factor

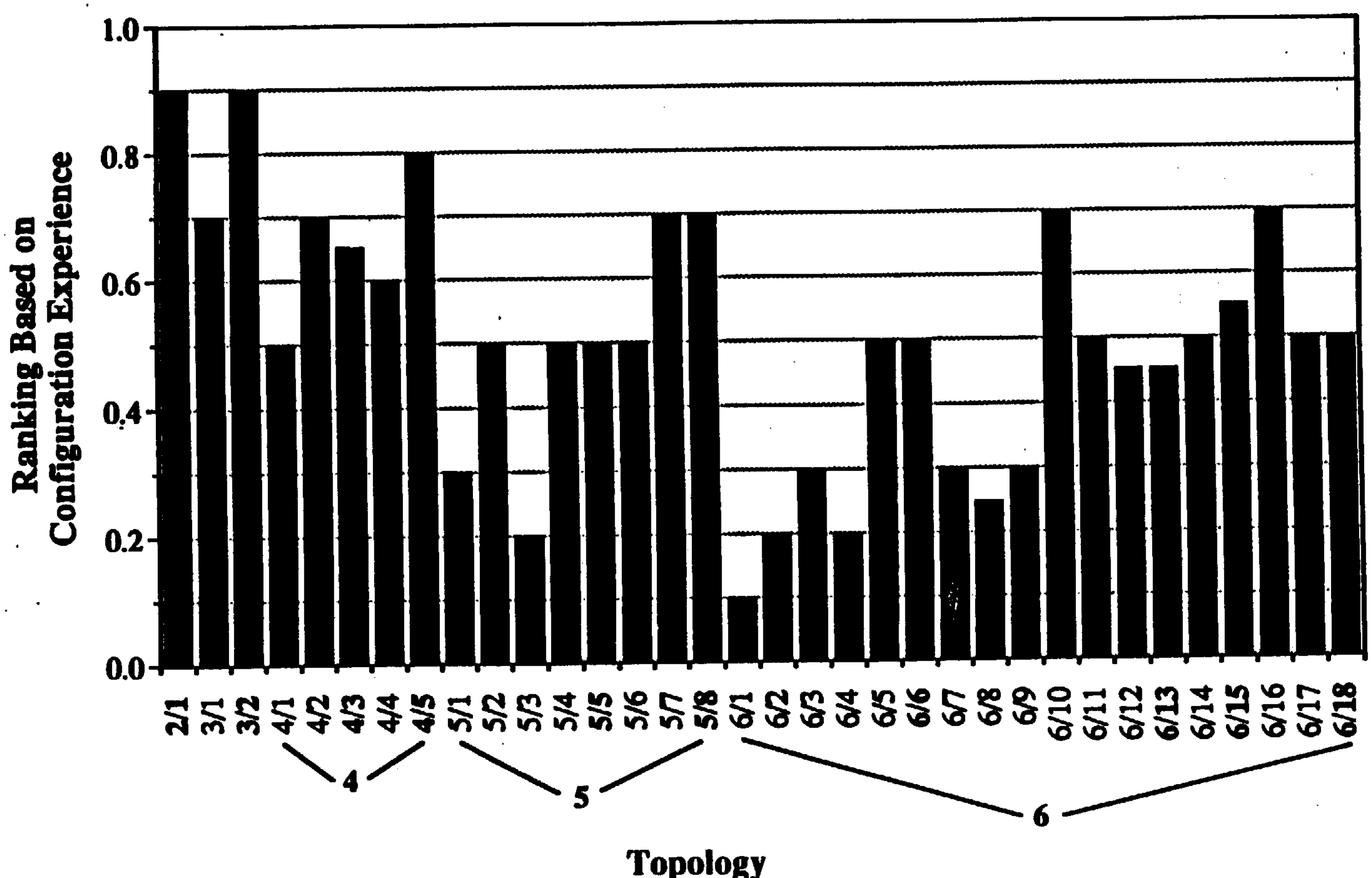
This is the most "subjective" of the metrics; however, its assessment was performed by configuration engineers who have developed contemporary aerobraked vehicle concepts over the last several years and are experienced with the configuration complications introduced by packaging behind aerobrakes. The goal here was to elucidate those topologies which, considered in cross section only (independent of module length), would facilitate configuring the smallest (and therefore lightest) aerobrake possible within each group of module-number.

Star and string configurations are poor; dense clusters, and particularly those which tend to acçommodate the curvature of an aerobrake shape and/or the conical aftbody wake-protection zone, trade much better. Selecting a cutoff (0.45 for example) allows rejecting the least favorable topologies.

Topology Metric Analysis

Aerobrake Integration Factor

$F_i = 0.1 - 0.9$



Topologies vary widely in their ability to be integrated easily behind aerobrakes.
The clusters are assessed regardless of module length.
Comparisons are most useful among clusters with the same number of modules.

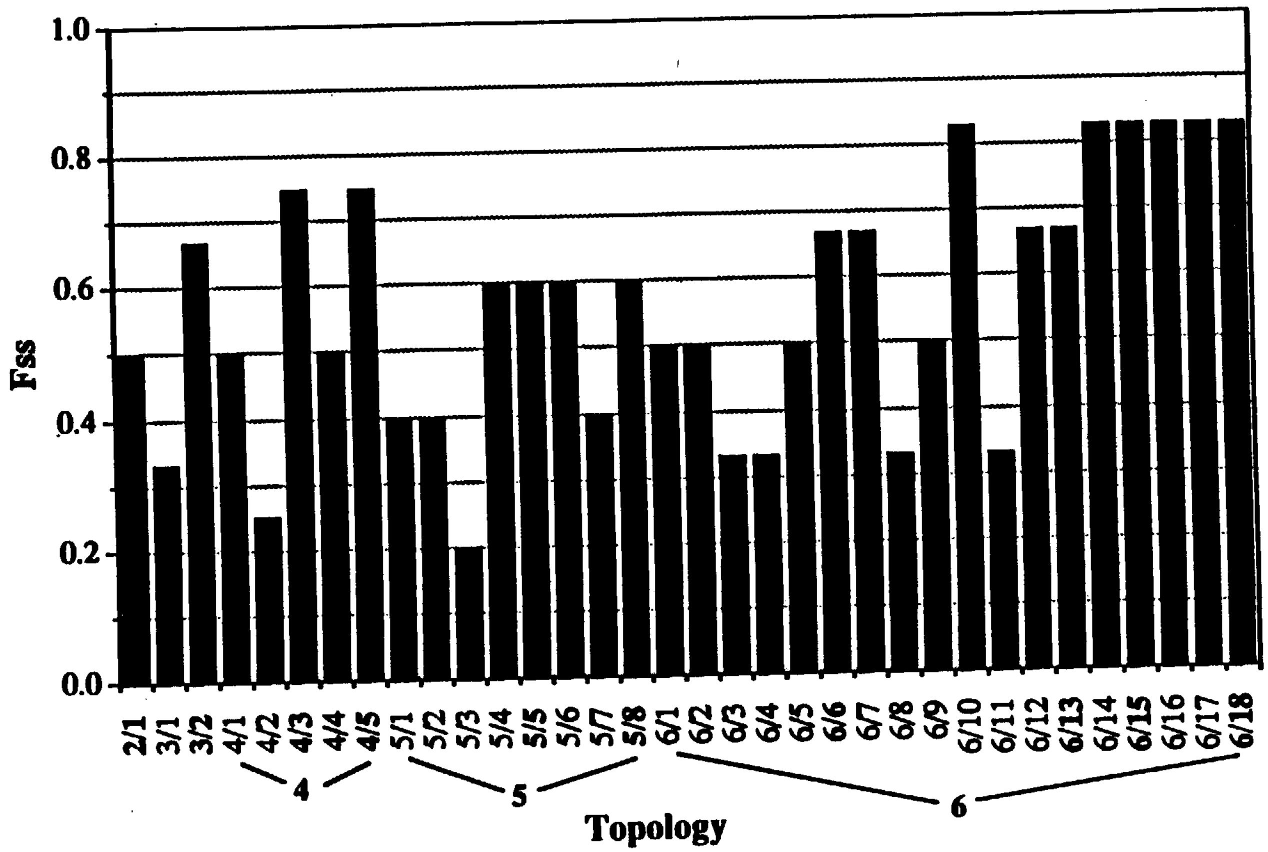
Topology Metric Analysis
Safe-Haven Split Factor

Assuming irrecoverable damage to a single module in flight, assuming individual modules cannot be moved within the topology after departure, and assuming the most favorable distribution of redundant ECLS equipment among the modules for each topology, this metric assesses how much of the original volume would be IVA-available to the crew for the remainder of the trip. Losing half of the total appears a severe scenario; such a criterion allows rejecting several topologies.

Topology Metric Analysis

Safe-Haven Split Factor

$$F_{ss} = \frac{n_{ss}}{n}$$



Requiring the worst-case safe-haven scenario to leave at least half the original habitable volume ($F_{ss} > 0.5$) eliminates many possible topologies

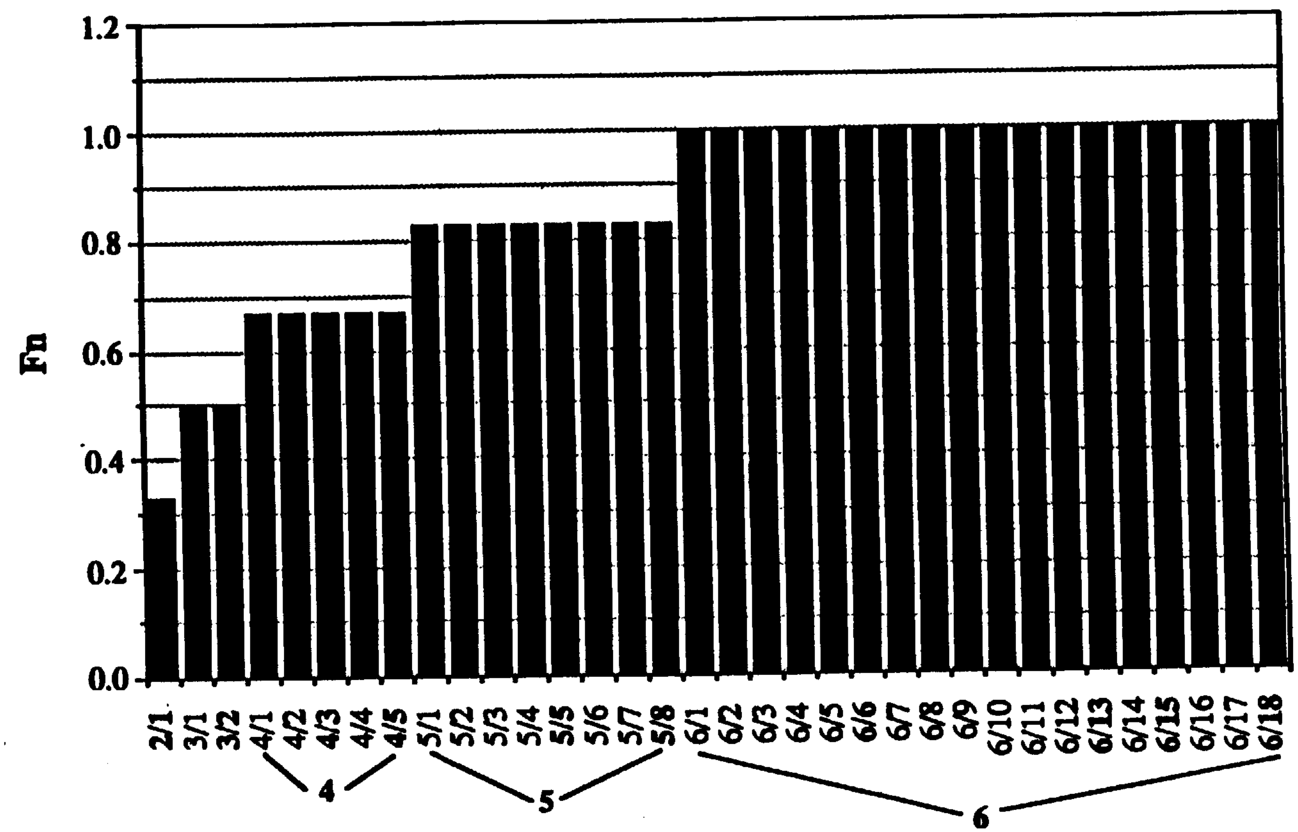
Topology Metric Analysis
Spatial Units Factor

This metric merely compares the total number of available modules to the maximum studied, six. It serves as a quantitative reminder that more separate modules provides more intrinsic opportunity for optimizing spatial units according to distinct functions (sleep, recreation, laboratory, etc.).

Topology Metric Analysis

Spatial Units Factor

$$F_n = \frac{n}{6}$$



Opportunities for spatial variety may be enhanced in clusters consisting of more modules

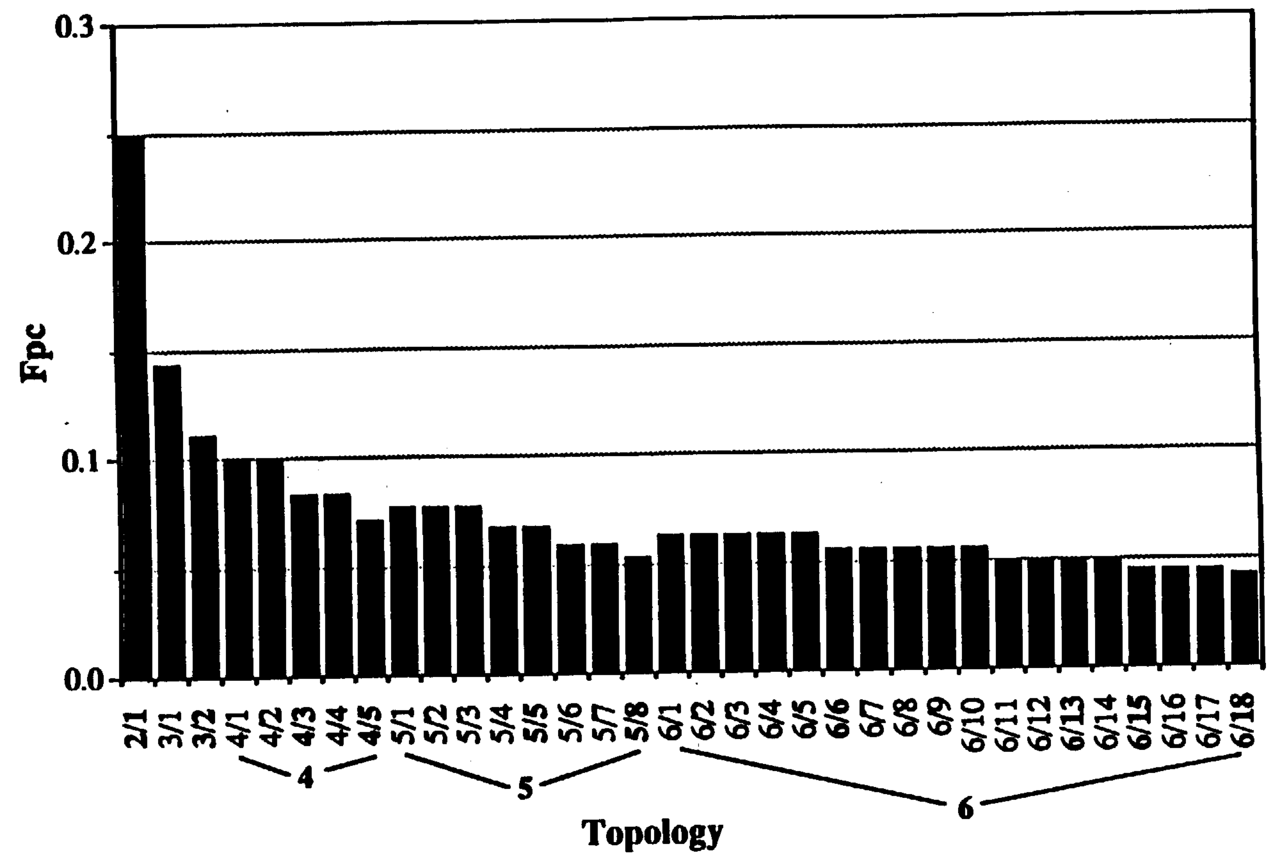
Topology Metric Analysis
Parts Count Factor

The greater the number of modules, however, the greater the parts count (tunnels, hatches, modules, interconnection structure, etc.), and the greater the opportunity for failures and leakage. The parts count metric drops dramatically once module-number exceeds 2 or 3. Subtle differences exist among topologies within each module-number group.

Topology Metric Analysis

Parts Count Factor

$$F_{pc} = \frac{1}{n + t}$$



The parts count increases quickly as topologies get more complex. However, the differential increase becomes less significant with larger numbers of modules

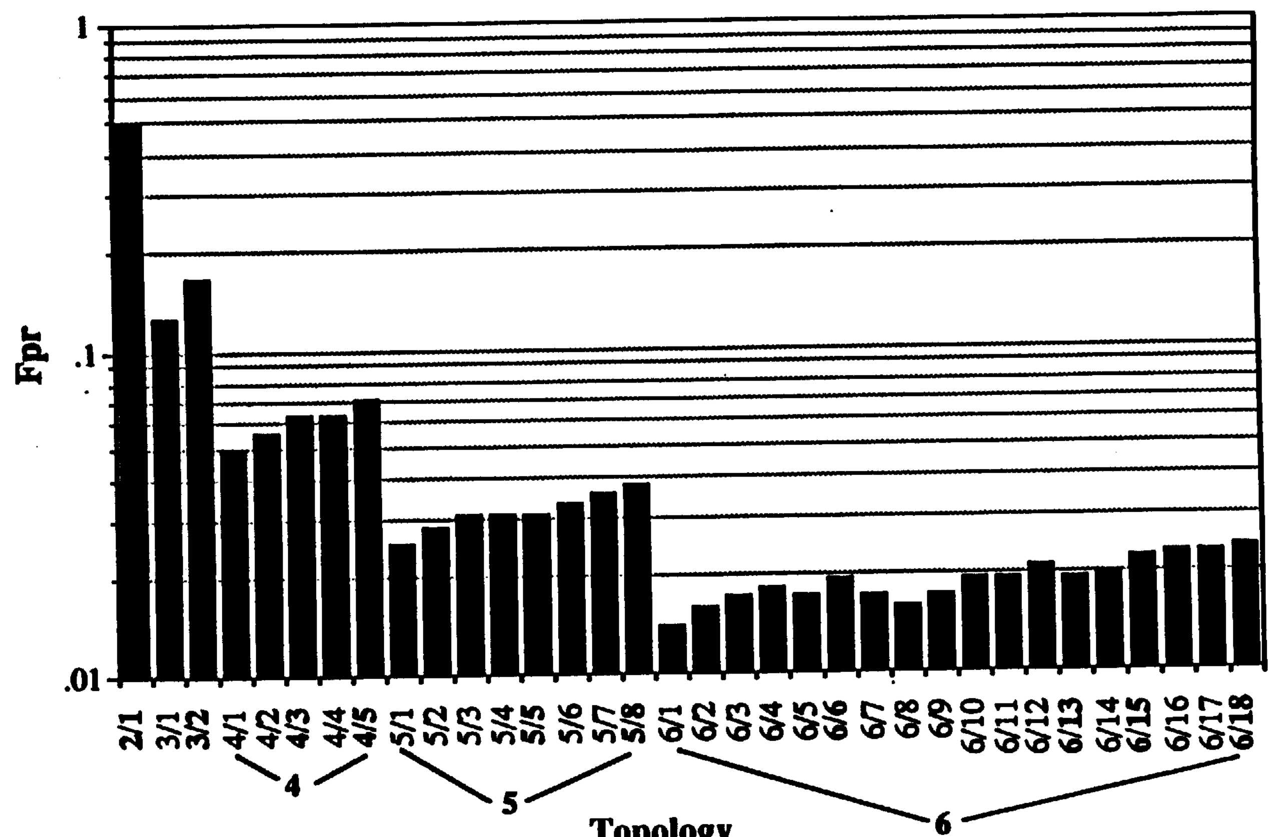
Topology Metric Analysis
Proximity Convenience Factor

This metric assesses how many non-destination modules one must go through to get to the destination module, summed in the best case over all possible combinations of origin and destination modules for all topologies. High numbers mean more convenient circulation, but low numbers may contribute to the perception of a greater habitable domain.

Topology Metric Analysis

Proximity Convenience Factor

$$F_{pr} = \sum_{i=1}^{n} \sum_{j=1}^{n-1} t$$



For high-n clusters, circulation patterns are characterized by having to thread several modules. This interferes with convenience, but contributes to perceptions of a large domain.

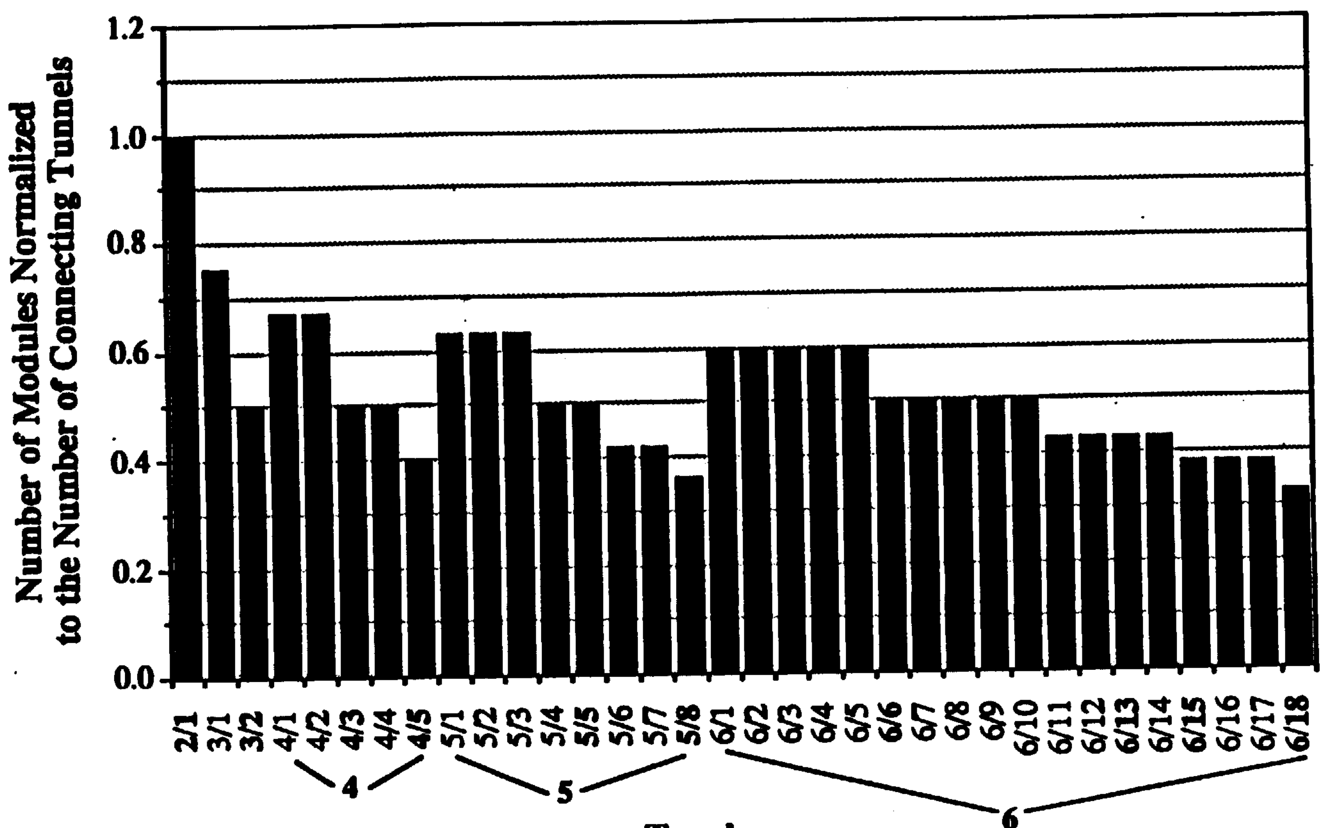
Topology Metric Analysis
Circulation Efficiency Factor

This metric merely ratios the number of modules to the number of tunnels used to connect them, for each topology. It is a measure of how much hardware is devoted to interconnection in the cluster approach to habitat design.

Topology Metric Analysis

Circulation Efficiency Factor

$$F_c = \frac{n}{t}$$



Options preferable for other reasons rarely tend to have the fewest connecting tunnels; however, suitable candidates can be selected from all cluster groups

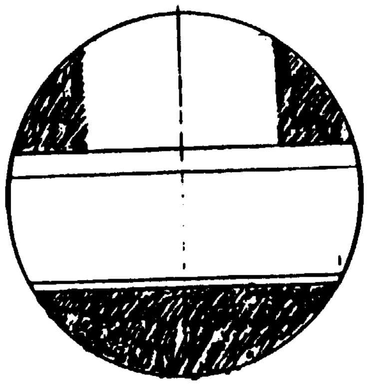
MTV Hab "Tunnel" Arrangements
Cross Section Properties

The next two charts show representative options for floor arrangements in the tunnel-oriented medium and large diameter modules. Off-nominal volume is defined as "uninhabitable", or that which has a ceiling height lower than the 2.3 m nominal assumed by this study; it is a prime location for equipment location. Lettered from left to right on the charts, options "B" and "C" provide the most nominal floor area, the most accessible underfloor volume (useful for ECLSS and stowage), and the advantages of the vaulted ceiling (spaciousness perception) without excessive wasted space. For the quantitative analysis purposes of this trade study, floor option B is selected.

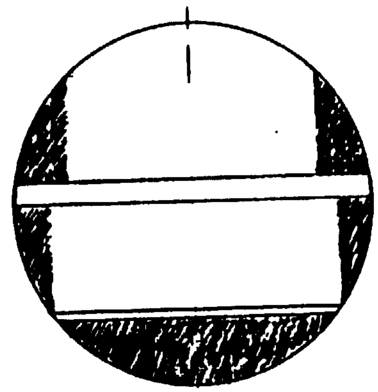
The third chart shows an analogous analysis for the small diameter module, for comparative purposes. For gravity conditions in which spaciousness is important for psychological reasons (long-duration flights), option "C" is most reasonable and is used for quantitative analyses throughout this trade study.

MTV Hab "Tunnel" Arrangements

7.6 m-diameter Cross Section Properties

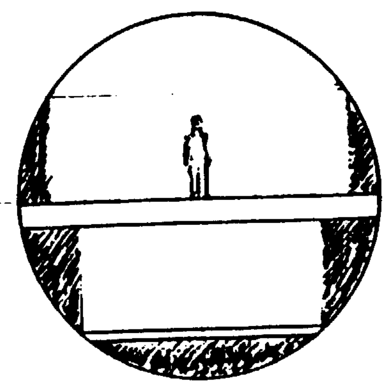


loor area: 10.8 m^2/m
ccessible off-nominal volume: 25 %
laximum ceiling height: 2.75 m
ut-of-reach overhead volume: 2 %

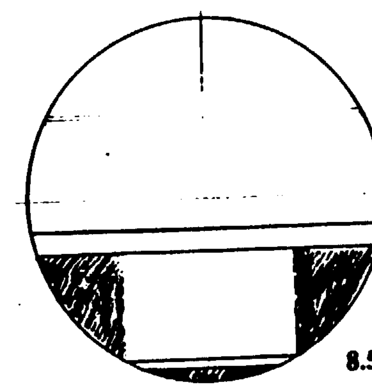


Floor area: 11.4 m^2/m
Accessible off-nominal volume: 22 %
Maximum ceiling height: 3.2 m
Out-of-reach overhead volume: 8 %

Floor area: 11.4 m^2/m
Accessible off-nominal volume: 17 %
Maximum ceiling height: 3.8 m
Out-of-reach overhead volume: 12 %



Floor area: 11.3 m^2/m
Accessible off-nominal volume: 12 %
Maximum ceiling height: 4.4 m
Out-of-reach overhead volume: 17 %



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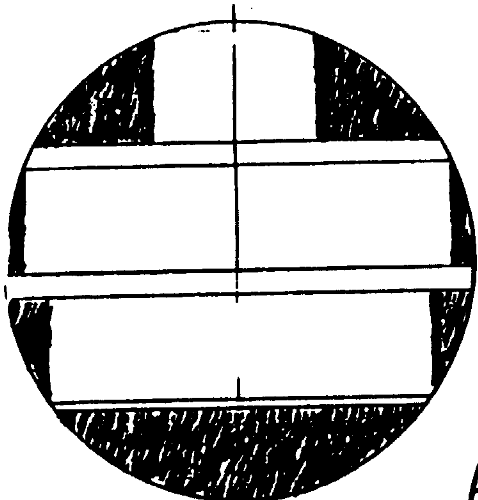
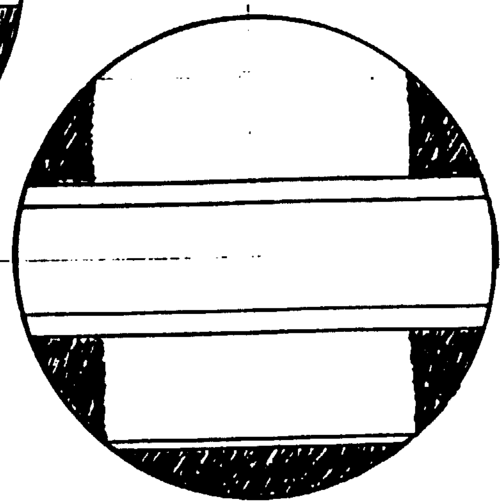
MTV Hab "Tunnel" Arrangements

10 m-diameter Cross Section Properties

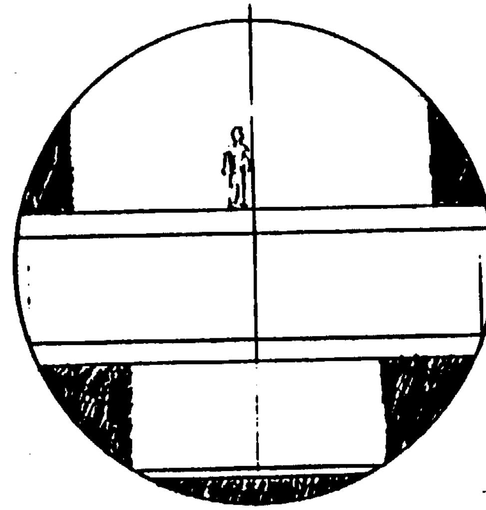


ɿloor area: 21.7 m^2/m
ɩccessible off-nominal volume: 25 %
ɬaximum ceiling height: 2.5 m
ʖut-of-reach overhead volume: 0 %

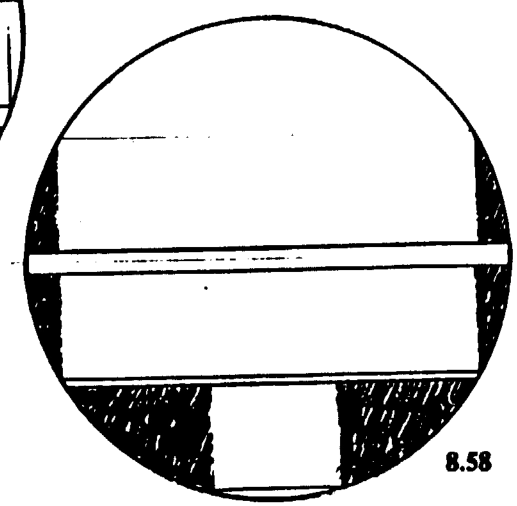


Floor area: 22.8 m^2/m
Accessible off-nominal volume: 15 %
Maximum ceiling height: 3.4 m
Out-of-reach overhead volume: 5 %

Floor area: 22.5 m^2/m
Accessible off-nominal volume: 12 %
Maximum ceiling height: 3.9 m
Out-of-reach overhead volume 9 %



Floor area: 21.3 m^2/m
Accessible off-nominal volume: 11 %
Maximum ceiling height: 4.75 m
Out-of-reach overhead volume: 19 %



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MTV Hab "Tunnel" Arrangements

4.4 m-diameter Cross Section Properties

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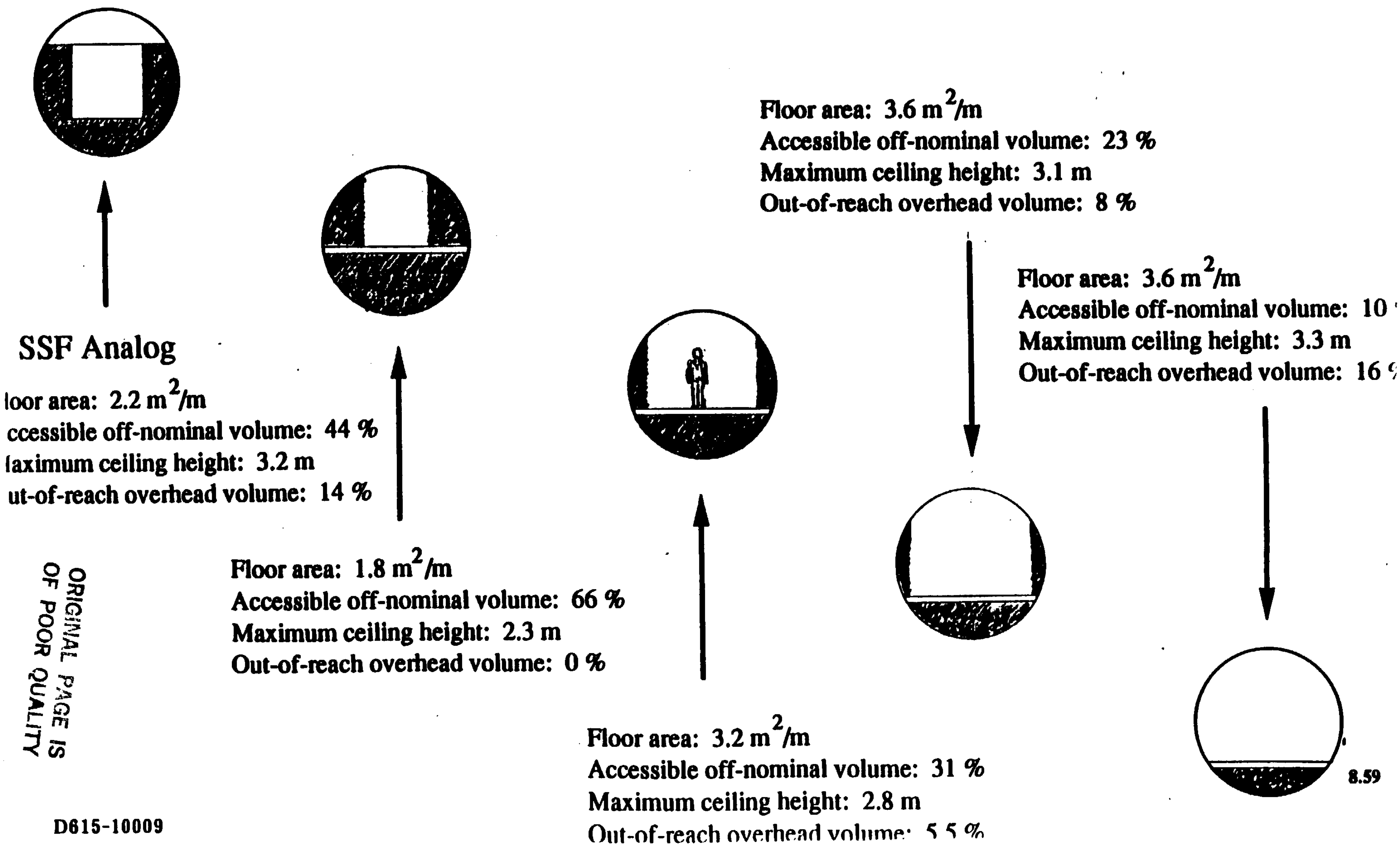

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Geometry Analysis Metrics

The next three charts define the metrics devised to compare the small module cluster options which survived their own neckdown, with the unitary medium and large diameter options. These have been developed to be configuration-independent; that is, they compare module geometries regardless of internal outfitting and detailing considerations. Each has been assigned a shorthand appellation, shown in quotation marks, which captures the primary way in which the metric has been taken for the purposes of this study.

Geometry Analysis Metrics

Nomenclature

A_n ≡ nominal floor area (having 2.3m ceiling height)
V ≡ volume
A_s ≡ sectional area of largest spatial unit
A_{soe} ≡ off-ergonomic sectional area (e.g. above 2.3m ceiling height)
l_{max} ≡ maximum simple path length within habitat
x ≡ plan dimension within one spatial unit
h ≡ maximum ceiling height
U ≡ number of spatial units
F ≡ number of floors
P_u ≡ spatial unit perimeter consumed by doorways to other spaces, and not available as wall space
O_d ≡ distinct pathways available between origin and destination spatial units
Spatial Unit ≡ 1 floor in multi-floor module, or 1 module in multi-module cluster

Guidelines

- All perception metrics are assessed independently of internal configuration details
- All metrics depend only on the geometry and orientation of the modules, and the arrangement of floors and spatial units within them

Symbol	Metric	Formula	Comments
F_{fa}	specific nominal floor area, "Inhabitability Factor"	$\frac{A_n}{\Sigma V}$	– Higher numbers mean more habitable floor area for gravity conditions. – Equipment can be located in off-nominal spaces or can take up nominal floor area (configuration-dependent)

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Geometry Analysis Metrics (2)

Symbol	Metric	Formula	Comments
F_s	Specific off-ergonomic section, "Vault Factor"	$\frac{A_{soe}}{\Sigma A_s}$	– Higher numbers indicate habitable spaces with more sectional area beyond the 2.3 m-high ergonomic envelope – Measures spaciousness in section
F_l	Specific end-to-end travel distance, "Domain Factor"	$\frac{l_{max}}{\Sigma V}$	– Higher numbers indicate long worst-case intra-habitat travel times – Lower numbers indicate habitats perceived as having limited territory
F_{rp}	Plan aspect ratio, "Hallway Factor"	$\frac{X_{max}}{X_{min}}$	– Taken in longest spatial unit – Higher numbers indicate more hallway-like spatial units
F_{rs}	Sectional aspect ratio, "Spaciousness Factor"	$\frac{X_{max} X_{min}}{h}$	– Taken in longest spatial unit – High numbers indicate perceptions of low ceiling height – Low numbers may indicate perceptions of being in a "pit"
F_u	Specific number of spatial units, "Variety Factor"	$\frac{U}{\Sigma V}$	– Higher numbers indicate more optimistic opportunitites to optimize different spaces

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Geometry Analysis Metrics (3)

Symbol	Metric	Formula	Comments
F_f	Specific number of floors, "Elevator Factor"	$\frac{F}{\sum A_n}$	- For small crew sizes, higher numbers indicate more "upstairs-downstairs" variety. - For large crew sizes in gravity configurations, higher numbers indicate functional inconvenience
F_p	Specific useful perimeter, "Perimeter Factor"	$\frac{\sum P_u}{\sum A_n}$	- Higher numbers indicate more intrinsically available wall space for equipment positioning
F_o	Options to destination "Pathway Factor"	$\sum_{U_i} \sum_{U_j} O_d$ $i,j = 1,\ldots n$ $i \neq j$	- Measures distinct pathways available within habitat system - Higher numbers indicate many options (can get extremely high) - Low numbers indicate monotony of movement patterns within the environment, deficiency of variety
F_v	Volume variety range, "Scale Factor"	$\frac{V_{max}}{V_{min}}$	– Measures the range of perceptual scales available to crew - Higher numbers indicate a wider range

Habitation Module Geometry Metrics

The following two charts tabulate the 10 geometry metrics for the options designated, as calculated according to the formulas just defined on the previous charts. An end-dome ellipsoid ratio of 3 was used for the medium and large diameter option calculations; 2 was used for the small-diameter options.

Habitation Module Geometry Metrics

Perception Metrics / Module Type	F_{fa}	F_s	F_l	F_{rp}	F_{rs}	F_u	F_f	F_p	F_o	F_v
4S-2/1	0.21	0.16	0.05	3.58	14.51	0.005	0.011	0.78	4	27.25
4M3-h	0.35	0.14	0.02	1.0	20.63	0.008	0.022	0.53	6	18.68
4M3-IB	0.20	0.25	0.02	1.24	11.69	0.005	0.026	0.74	4	13.73
4L3-h	0.34	0.23	0.02	1.0	31.25	0.004	0.013	0.40	2	37.57
4L3-IB	0.18	0.27	0.02	1.83	6.99	0.007	0.037	0.87	16	9.00
6S-2/1	0.22	0.16	0.04	5.56	22.50	0.003	0.007	0.70	4	40.88
6S-3/2	0.25	0.16	0.04	4.22	17.10	0.005	0.012	0.75	24	27.25
6M3-h	0.36	0.14	0.02	1.0	20.63	0.008	0.022	0.53	6	18.68
6M3-IB	0.22	0.25	0.02	2.22	20.97	0.003	0.014	0.57	4	24.63
6L3-h	0.35	0.23	0.02	1.0	31.25	0.005	0.013	0.40	6	37.57
6L3-IB	0.21	0.27	0.02	1.08	12.23	0.005	0.021	0.61	16	15.75
8S-2/1	0.22	0.16	0.04	7.56	30.60	0.002	0.005	0.66	4	54.50
8S-3/2	0.22	0.16	0.03	4.92	19.91	0.003	0.011	0.72	24	36.33
8S-4/3	0.21	0.16	0.03	3.58	14.51	0.005	0.011	0.78	112	27.25
8S-4/5	0.21	0.16	0.02	3.58	14.51	0.005	0.014	0.78	148	27.25
8M3-h	0.36	0.14	0.02	1.0	20.63	0.008	0.022	0.53	30	18.68
8M3-IB	0.23	0.25	0.02	2.71	25.61	0.003	0.012	0.53	4	30.08
8L3-h	0.36	0.23	0.01	1.0	31.25	0.005	0.013	0.40	12	37.57
8L3-IB	0.23	0.27	0.02	1.36	17.47	0.003	0.015	0.51	16	22.50
10S-3/2	0.22	0.16	0.03	6.22	25.20	0.003	0.008	0.68	24	45.42

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Habitation Module Geometry Metrics

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Module Type \ Perception Metrics	F_{fa}	F_s	F_l	F_{rp}	F_{rs}	F_u	F_f	F_p	F_o	F_v
10S-4/3	0.22	0.16	0.03	4.58	18.56	0.004	0.008	0.73	112	34.06
10S-4/5	0.22	0.16	0.02	4.58	18.56	0.004	0.012	0.73	148	34.06
10S-5/5	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	224	27.25
10S-5/8	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	660	27.25
10M3-h	0.37	0.14	0.02	1.0	20.63	0.008	0.022	0.53	56	18.68
10M3-IB	0.23	0.25	0.02	3.69	34.89	0.002	0.009	0.48	4	40.98
10L3-h	0.36	0.23	0.01	1.0	31.25	0.005	0.013	0.40	116	37.57
10L3-IB	0.24	0.27	0.02	1.77	24.13	0.003	0.011	0.45	16	29.25
12S-3/2	0.22	0.16	0.03	7.56	30.60	0.002	0.007	0.66	24	54.50
12S-4/3	0.22	0.16	0.03	5.56	22.50	0.003	0.007	0.70	112	40.88
12S-4/5	0.22	0.16	0.02	5.56	22.50	0.003	0.009	0.70	148	40.88
12S-5/5	0.22	0.16	0.03	4.36	17.66	0.004	0.007	0.74	224	32.70
12S-5/8	0.22	0.16	0.02	4.36	17.66	0.004	0.007	0.74	660	32.70
12S-6/6	0.21	0.16	0.03	3.58	14.51	0.005	0.007	0.78	392	27.25
12S-6/13	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	800	27.25
12S-6/15	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	1,392	27.25
12S-6/18	0.21	0.16	0.02	3.58	14.51	0.005	0.013	0.78	1,988	27.25
12M3-h	0.37	0.14	0.02	1.0	20.63	0.008	0.022	0.53	72	18.68
12M3-IB	0.23	0.25	0.02	4.18	39.53	0.002	0.008	0.47	4	46.43
12L3-h	0.36	0.23	0.01	1.0	31.25	0.005	0.013	0.40	208	37.57
12L3-IB	0.25	0.27	0.02	2.18	27.95	0.002	0.009	0.42	16	36.00

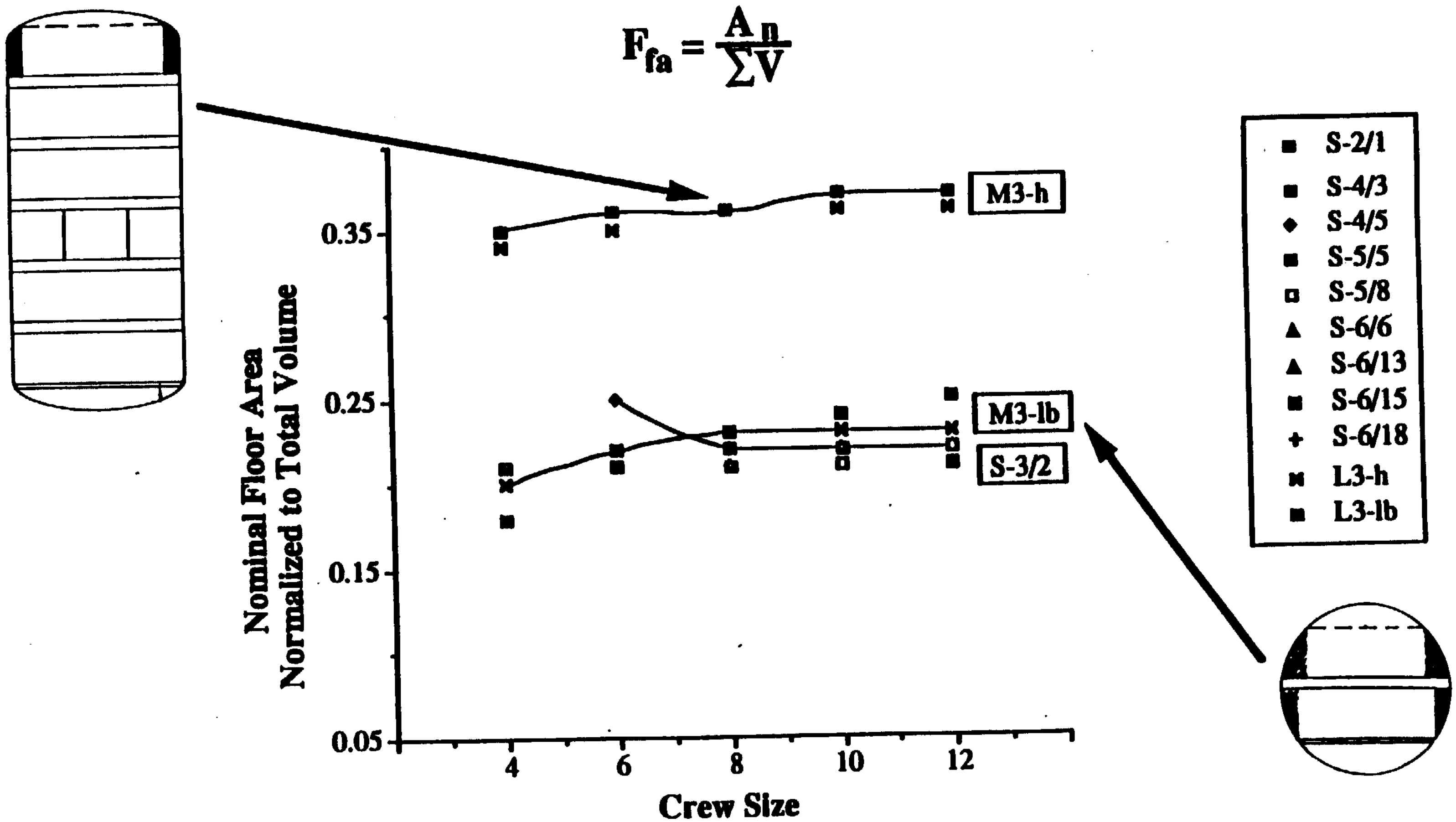
Geometry Metric Analysis
"Inhabitability Factor"

This metric assesses how much of the floor area has a ceiling height of at least 2.3 m (and is therefore nominal by our definition), relative to the total habitat volume. It quantifies the familiar result that walls which curve vertically introduce greater habitability penalties than walls which are normal to the floor.

Geometry Metric Analysis

"Inhabitability Factor"

$$F_{fa} = \frac{A_n}{\sum V}$$



There is more specific nominal floor area in stacked configurations than in tunnel configurations, regardless of module diameter.

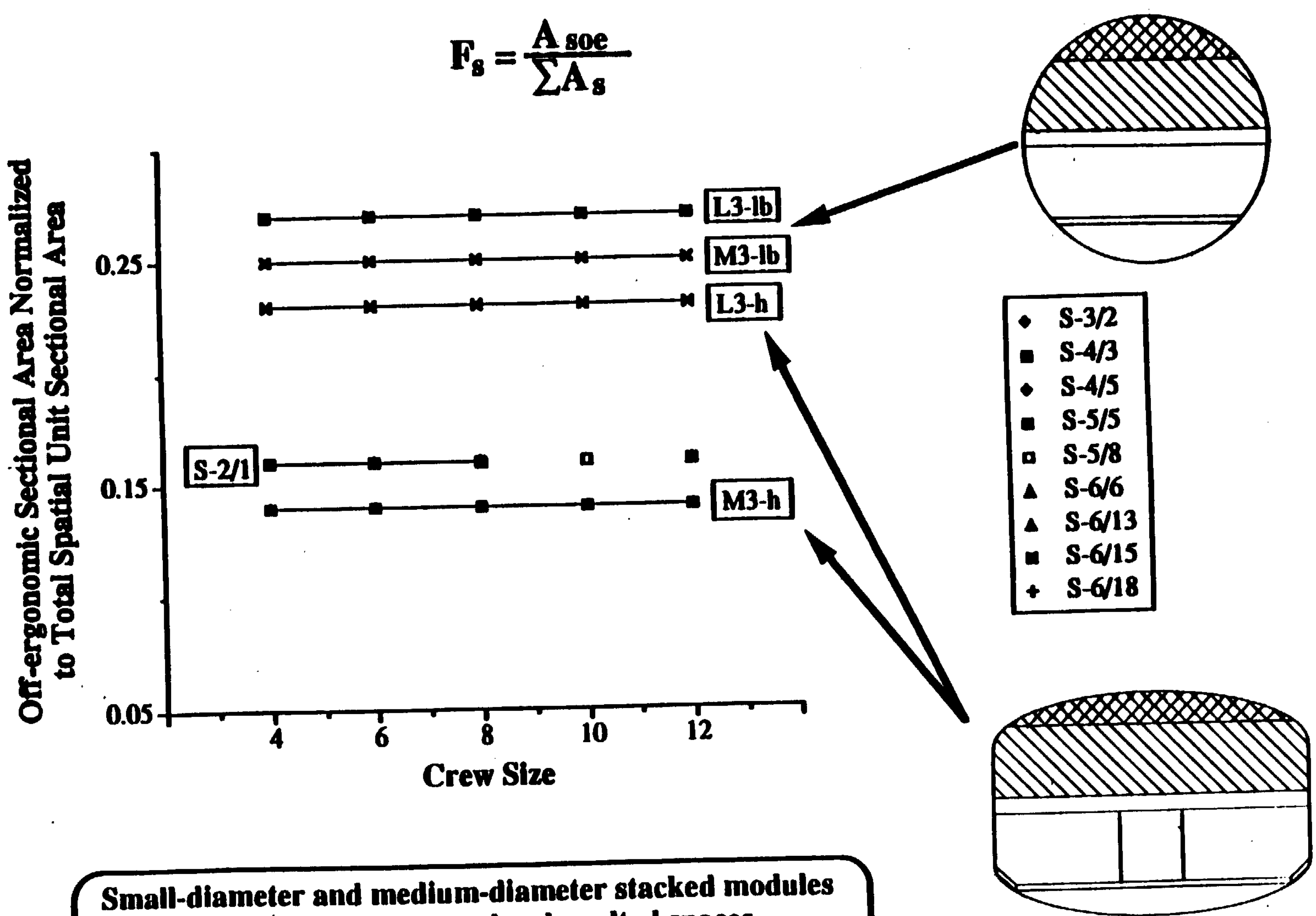
Geometry Metric Analysis
"Vault Factor"

This metric assesses, in the most spatially generous place within each option, the ratio of out-of-reach cross section (in a gravity field) to total cross section. It was developed to pertain to overhead vaults, but the measure remains similar even for μg conditions, since the human reach envelope travels with the body. An indication of how much "height" is out of reach at any time, this metric implies spaciousness in section. Large diameters trade best in both orientations, as does the tunnel orientation of the medium diameter (this result is sensitive to floor configuration assumptions).

Geometry Metric Analysis

"Vault Factor"

$$F_s = \frac{A_{soe}}{\sum A_s}$$



Small-diameter and medium-diameter stacked modules have less generous overhead vaulted spaces.

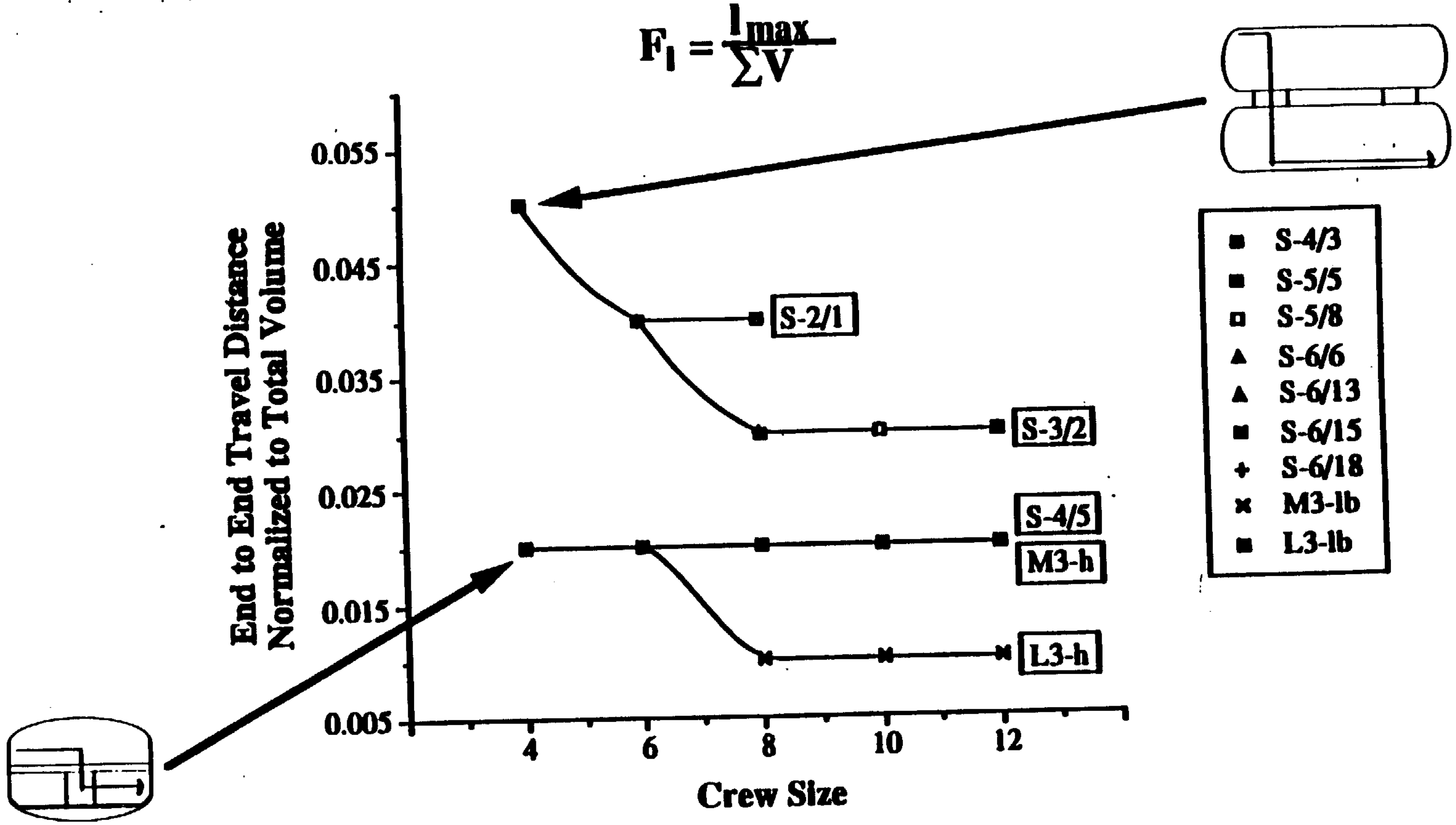
Geometry Metric Analysis
"Domain Factor"

This metric assesses the travel distance from one "end" of the habitat system to the most distal point, normalized to total volume. It is taken as a measure of the domain available in the confined habitat, since long travel times may imply more inhabited territory. However, longer travel times also introduce greater locomotion delays in an emergency. The medium and large diameter options have more compact domains.

Geometry Metric Analysis

"Domain Factor"

$$F_l = \frac{l_{max}}{\sum V}$$



Small-diameter modules would seem to cover more territory, but larger diameter modules allow shorter emergency access times.

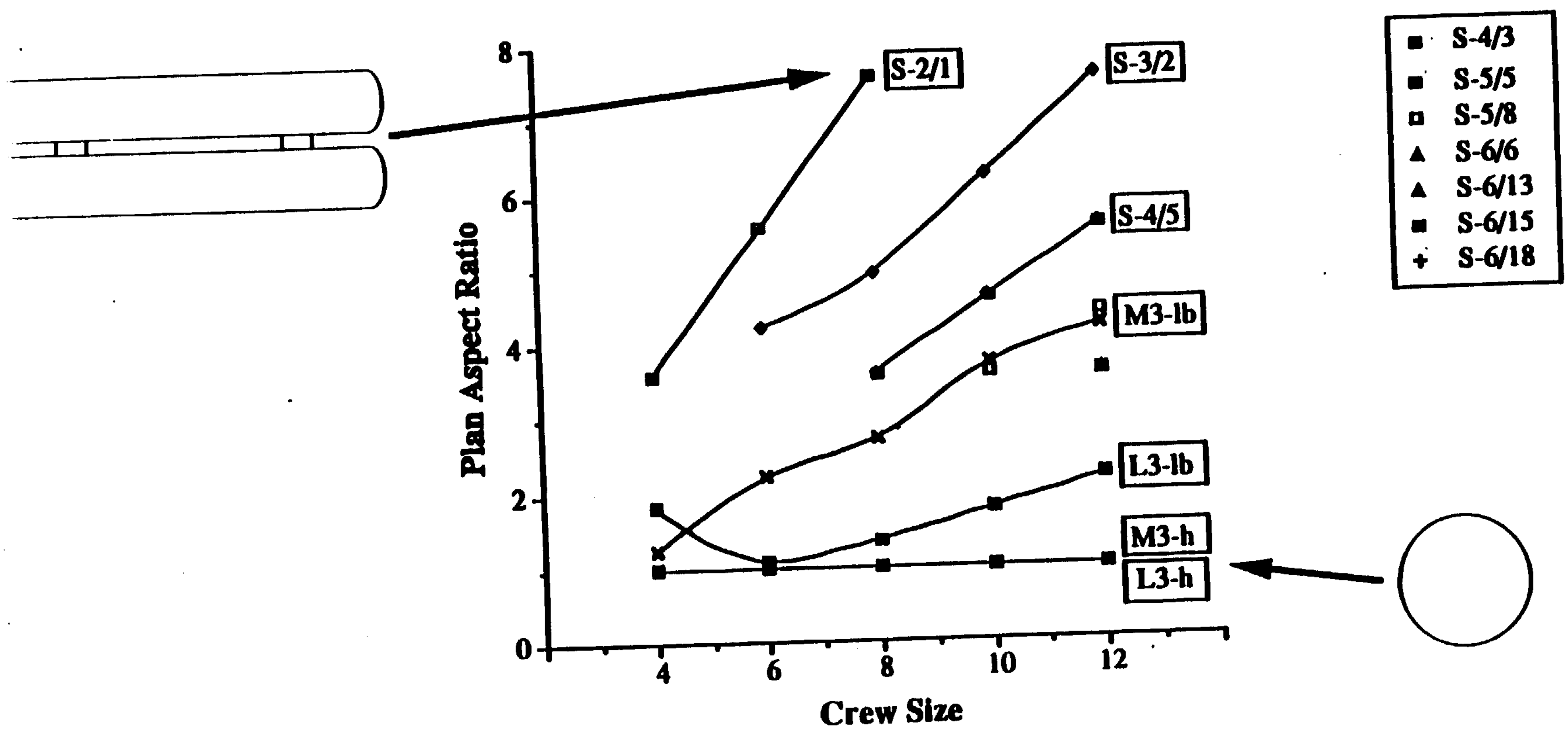
Geometry Metric Analysis
"Hallway Factor"

This metric assesses aspect ratio in plan of the largest single perceivable spatial unit within the habitat (one module in a cluster, or one floor in unitary options). The stacked options remain constant with increasing crew size, because the dimensions per floor remain constant. The tunnel options all become more like hallways with increasing crew size, since they get longer in plan. For the small diameter options, the fewer modules in the cluster, the longer each must become with larger crew sizes, and therefore the steeper the slope of the curve.

Geometry Metric Analysis

"Hallway Factor"

$$F_{rp} = \frac{x_{max}}{x_{min}}$$



Medium and large diameter options provide more evenly-proportioned spatial units.

Geometry Metric Analysis
"Spaciousness Factor"

This metric assesses sectional aspect ratio of the largest perceivable volume within each option. This is a more apt measure of overall spaciousness than the "vault factor", because it includes three dimensions. Stacked options remain constant with increasing crew size, because the top floor does not change dimension. The dome they provide trades very well for the smallest crew sizes, but is passed by the tunnel options for the larger crew sizes because their barrel vaults grow in length commensurately.

Geometry Metric Analysis

"Spaciousness Factor"

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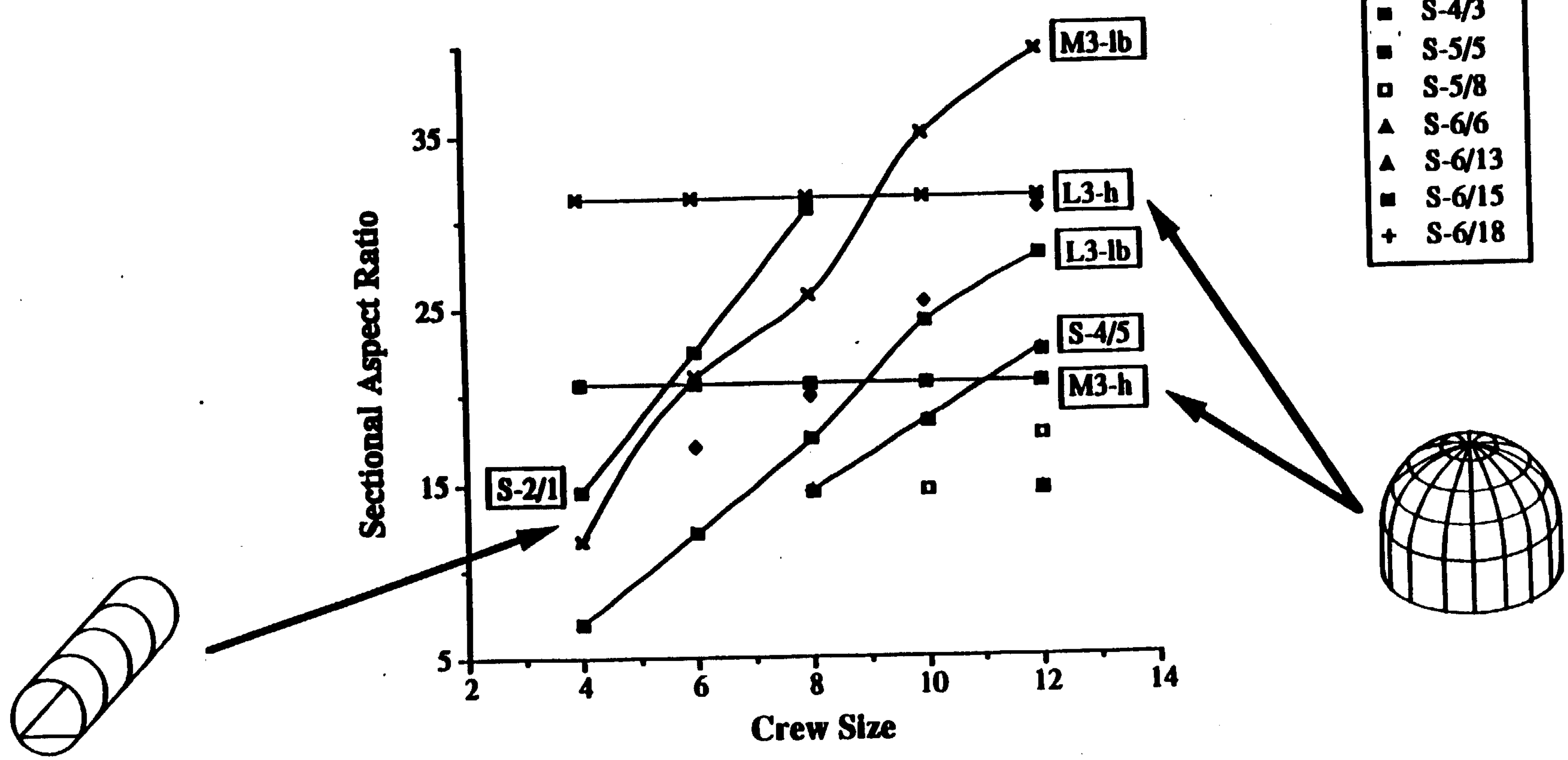
$$F_{rs} = \frac{(x_{max})(x_{min})}{h}$$



Domed spaces appear more spacious for small crew sizes. With fixed diameter, barrel-vaulted spaces trade better for larger crew sizes.

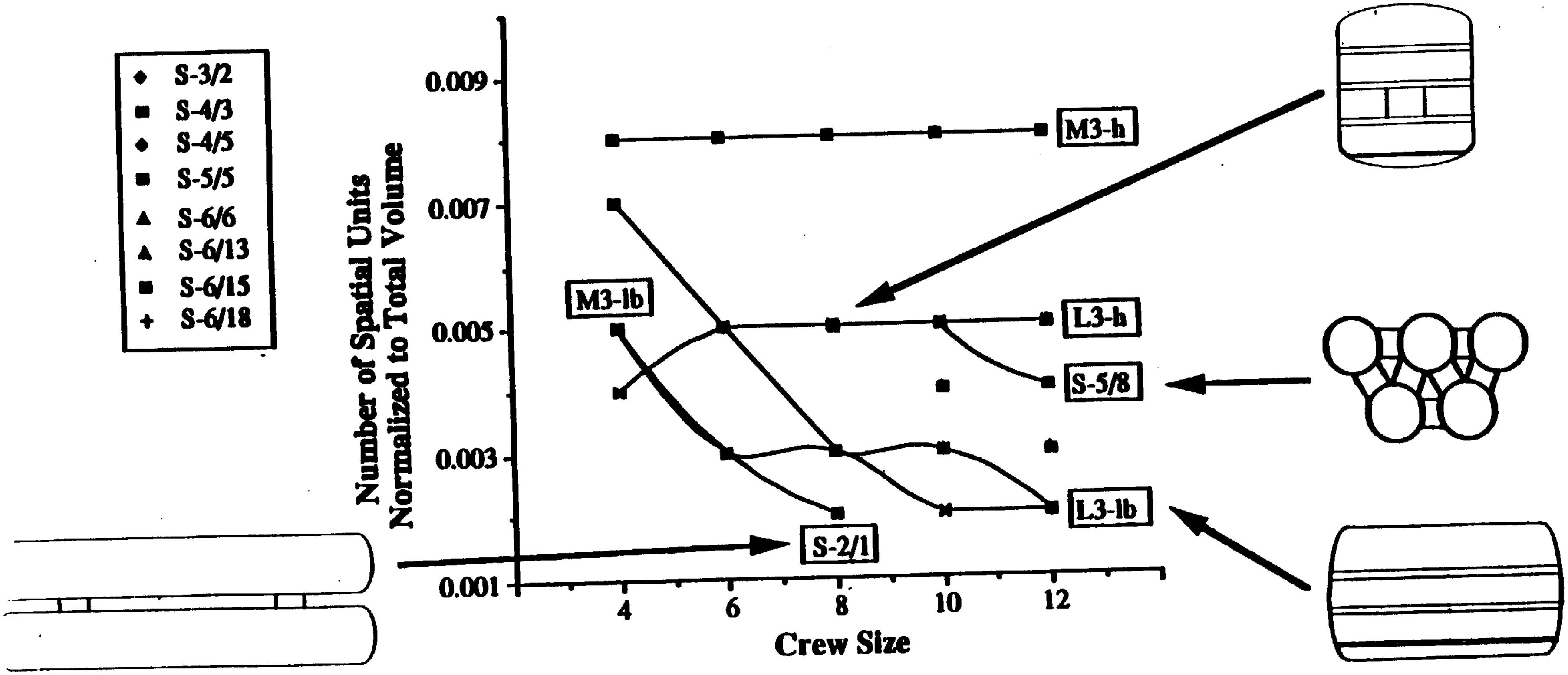
Geometry Metric Analysis
"Variety Factor"

This metric assesses how many perceptual pieces the available volume is broken up into, which measures (among other things) how much opportunity exists for optimizing spatial units for distinct functions. It is similar to the Spatial Units Factor in the Topology Metric Analysis, but includes the larger diameter options as well, and is normalized to total volume. The tunnel options trade poorly for large crew sizes (although interior designs could generate more spatial units with the cavernous volume available).

Geometry Metric Analysis

"Variety Factor"

$$F_u = \frac{U}{\sum V}$$



Stacked, and most cluster, configurations provide more opportunities for functionally unique spatial units than do tunnel configurations.

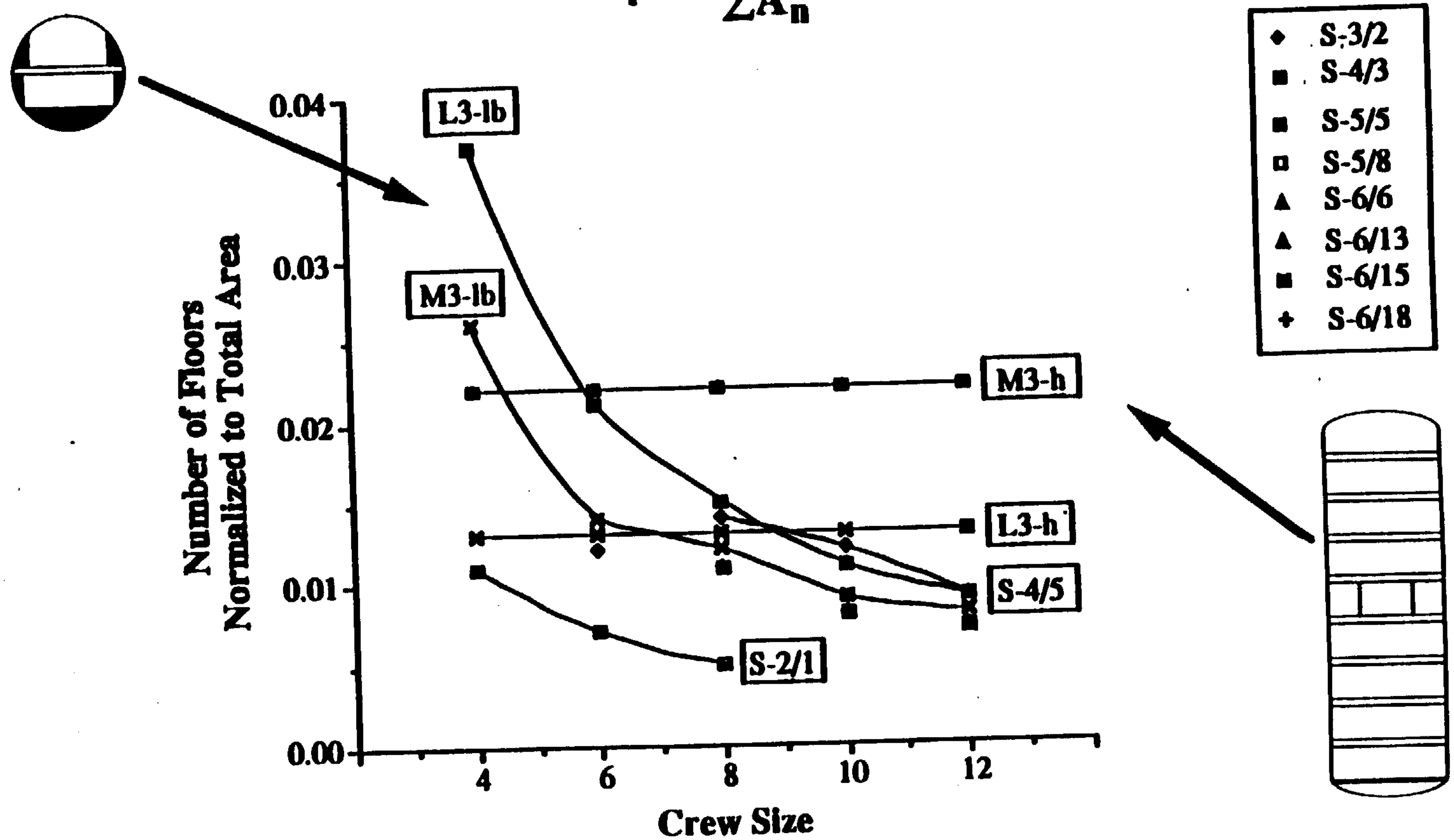
Geometry Metric Analysis
"Elevator Factor"

This metric reveals how many separate floors the available floor area is broken up into. "Split-level" cluster options were assigned fractional numbers-of-floors for the calculations. Whereas the metric may be largely irrelevant for μg conditions, in a gravity field it provides a strong discriminator against particularly the stacked medium diameter option for large crew sizes (analogous to living in a 9-story house with one room on each floor). For th smallest w sizes, the medium and large modules get so short that the tunnel orientation becomes a less efficient was to organize the internal space.

Geometry Metric Analysis

"Elevator Factor"

$$F_f = \frac{F}{\sum A_n}$$



Stacked options for large crew sizes have an excessive number of floors in gravity orientations. Medium and large diameter tunnel options work inefficiently for small crew sizes.

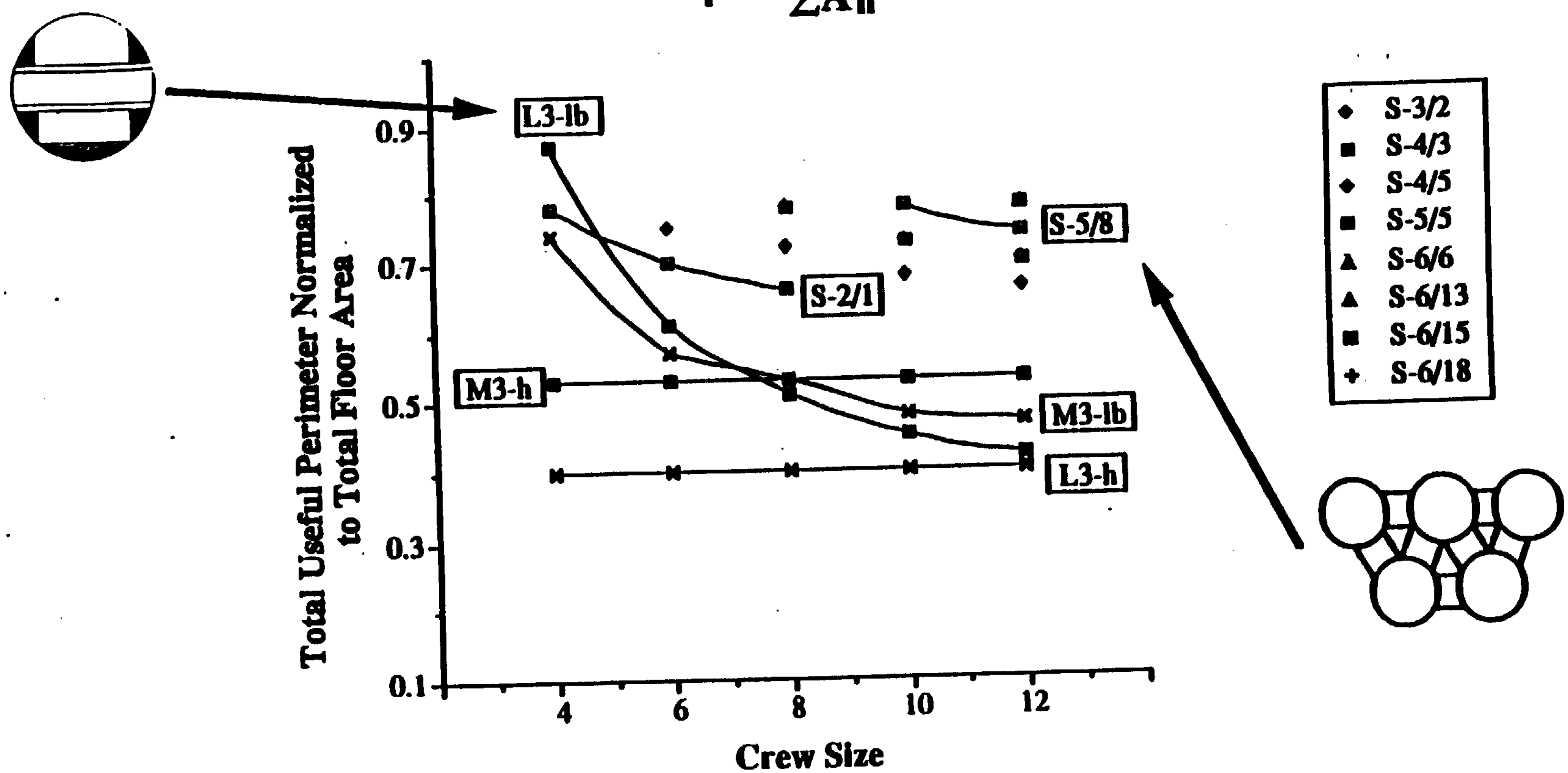
Geometry Metric Analysis
"Perimeter Factor"

This metric was devised to investigate the penalty in usable "room" perimeter suffered by module cluster options sporting a lot of interconnection tunnels. What it reveals, however, is that the small diameter options have so much more specific surface area that the tunnel effect washes out; the larger diameter options have much less intrinsic wall area available. This means that equipment mounting cannot as readily take advantage of wall locations for these latter options; however, their reduced pressure vessel wall area will be seen to confer a mass advantage.

Geometry Metric Analysis

"Perimeter Factor"

$$F_p = \frac{\sum P_u}{\sum A_n}$$



Although there are modest differences among cluster options due to tunnel openings, these modules provide more inherent wall space than larger diameter options because of their greater surface area.

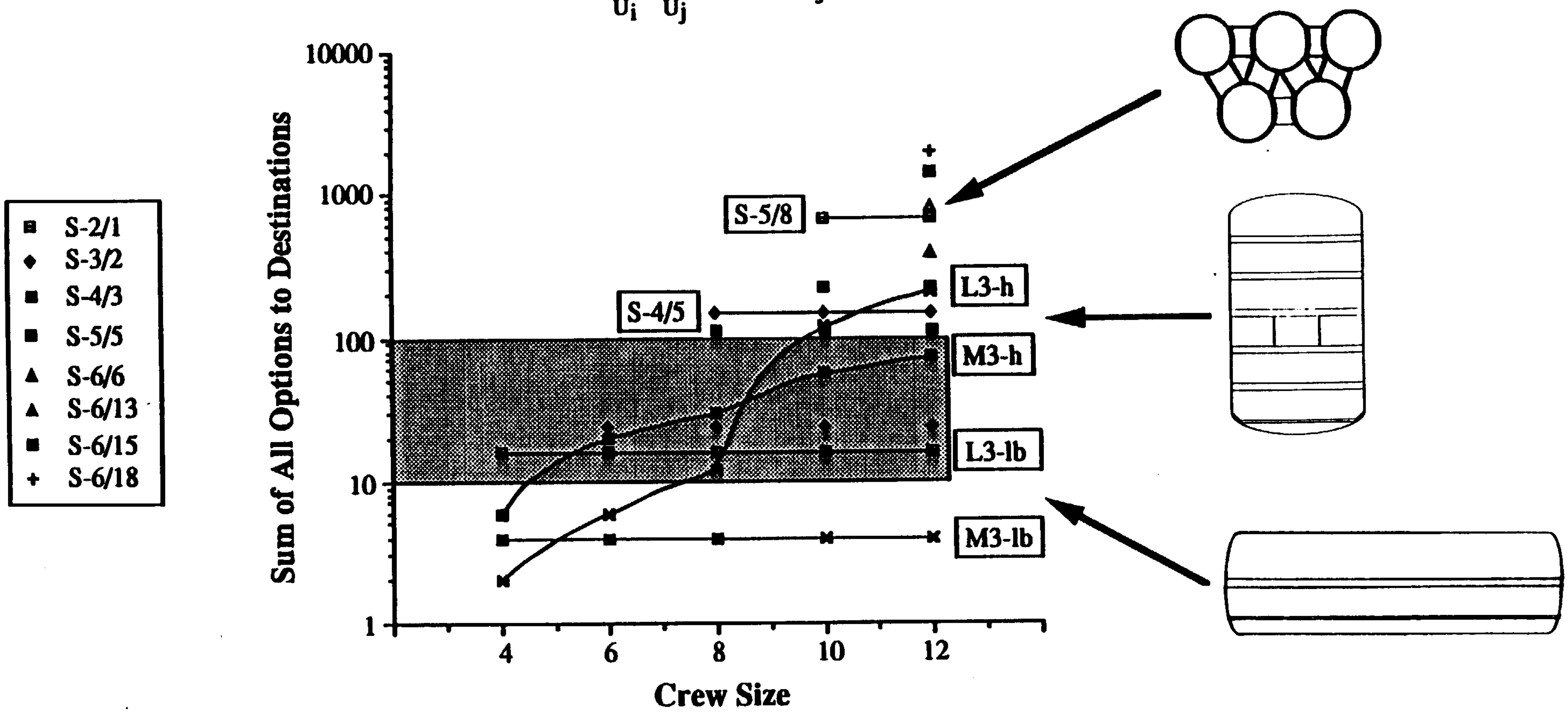
Geometry Metric Analysis
"Pathway Factor"

This metric calculates the number of different ways to get from one spatial unit to another in the habitat options, summed over all combinations of starting and destination units. Note the logarithmic ordinate, which ranges from 2 to over 2000. The discontinuity in the curves for the larger diameter stacked options reveals the assumption than two separate vertical circulation paths are required for crew sizes larger than 8, just to avoid circulation congestion. Interconnected clusters can provide many, many pathway options. This may be quite advantageous in mitigating "domain boredom" over long durations, and in alleviating social concentration when undesirable. Perceptions of inherent habitat privacy accommodation may be enhanced with many pathway options. A suggested range of pathway numbers is indicated on the graph; because it has only two floors, the medium diameter tunnel option trades quite poorly.

Geometry Metric Analysis

"Pathway Factor"

$$F_o = \sum_{U_i} \sum_{U_j} O_d \left(\begin{smallmatrix} i,j = 1,\dots 6 \\ i \neq j \end{smallmatrix} \right)$$



The intermediate range numbers (~10 to 100) provide sufficient circulation options to mitigate boredom, without incurring an excessive hardware penalty.

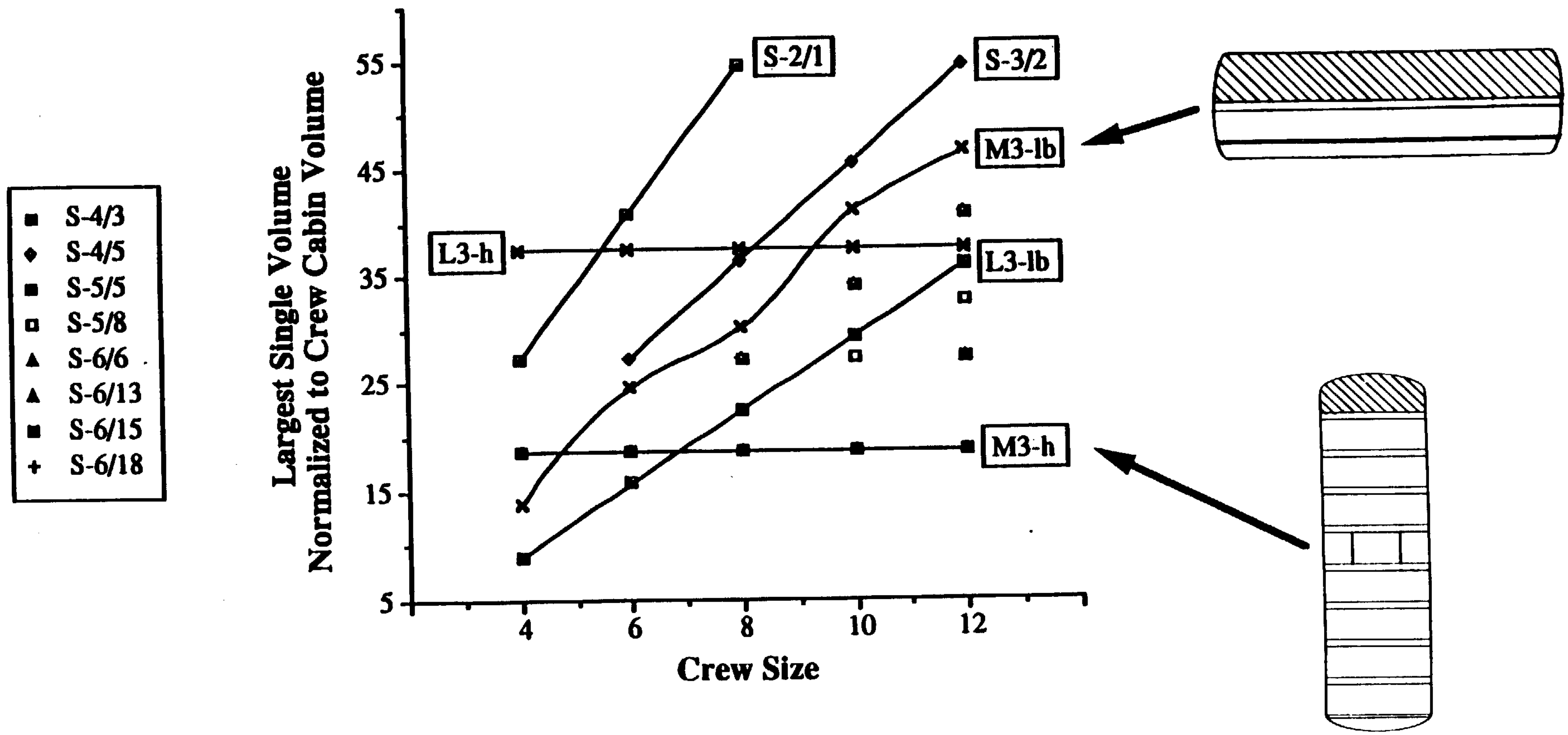
Geometry Metric Analysis
"Scale Factor"

This metric compares the largest single spatial unit available to an individual crew cabin, measured in volume, as an indication of the range of spatial scales available within each habitat option. A greater range may imply greater potential spatial variety when the interior is configured, important to mitigate perceptual boredom over long durations. For small crew sizes, the top floor dome of the large diameter stacked option is dramatic; for larger crew sizes, however, the tunnel options are more favorable. The seemingly good performance of some cluster options for large crew sizes shows that some of those modules get very long; this must be weighed against their large "Hallway Factor".

Geometry Metric Analysis

"Scale Factor"

$$F_v = \frac{V_{max}}{V_{min}}$$



With increasing crew size, tunnel options provide potential for greater variety of spaces.

Activity & Proximity Analysis

In addition to the topology and geometry metric analyses, it is important to determine if any unique complications arise from the interior configuration standpoint for the primary habitat module options.

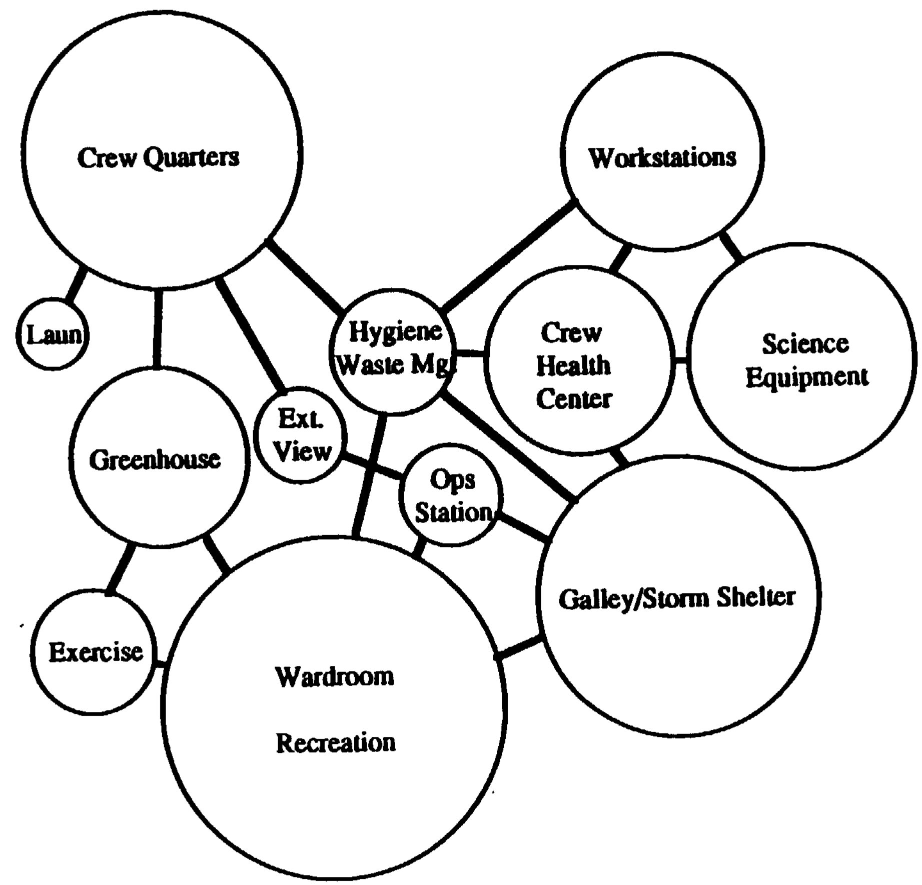
Using SSF and terrestrial design as starting points, we developed representative functional area allocations for habitat activities (the area bias emphasizes the gravity condition --- a μg bias would emphasize packing volume as well as surface area). The allocations listed are totals per crew of four. Excepting those values noted as "per crew", which remain constant to first order despite crew size changes, the relative areas scale with increased crew size. The activities taking place in those allocated areas are related by proximity constraints of varying strengths, to be close together or far apart. For example, recreation activities should be far from sleep areas to avoid disturbing resting crew members. However, most habitation areas and the recreation area should have viewing access to greenhouse facilities. The proximity diagram then serves as a guide for developing interior configurations which satisfy functional and perceptual requirements.

Activity & Proximity Analysis

Representative Allocations
(referenced as floor area for gravity configuration)

Activity Location	Area (m^2)
Crew Quarters	16
PH/WMF	3
Galley/Storm Shelter	16
Wardroom/Recreation	12
Exercise	2
Greenhouse	6
Operations Station	2 (per crew)
Workstations	8
Science Equipment	10
CHC	7
ECLSS	12
EVA Stowage	2
Laundry	1 (per crew)
Spares Stowage	7 (15% active equip)
Circulation	9 (15% crew space)

Scaled Proximity Diagram



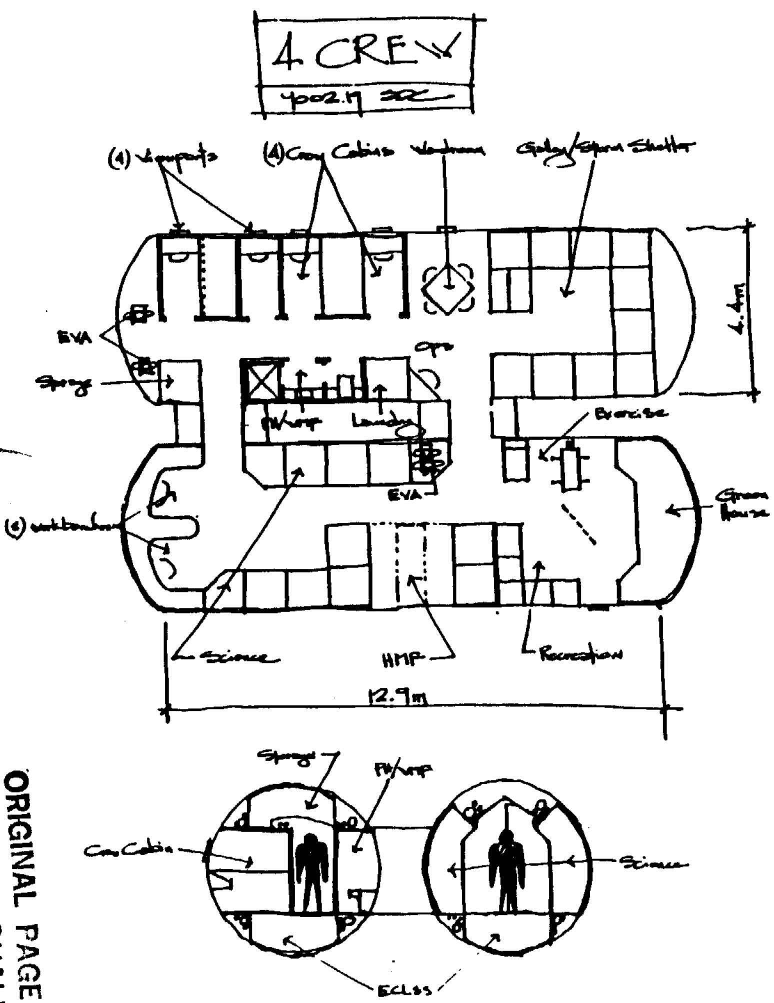
The proximity analysis provides the basis for developing interior configurations which work and which address the unique requirements of long-duration spaceflight

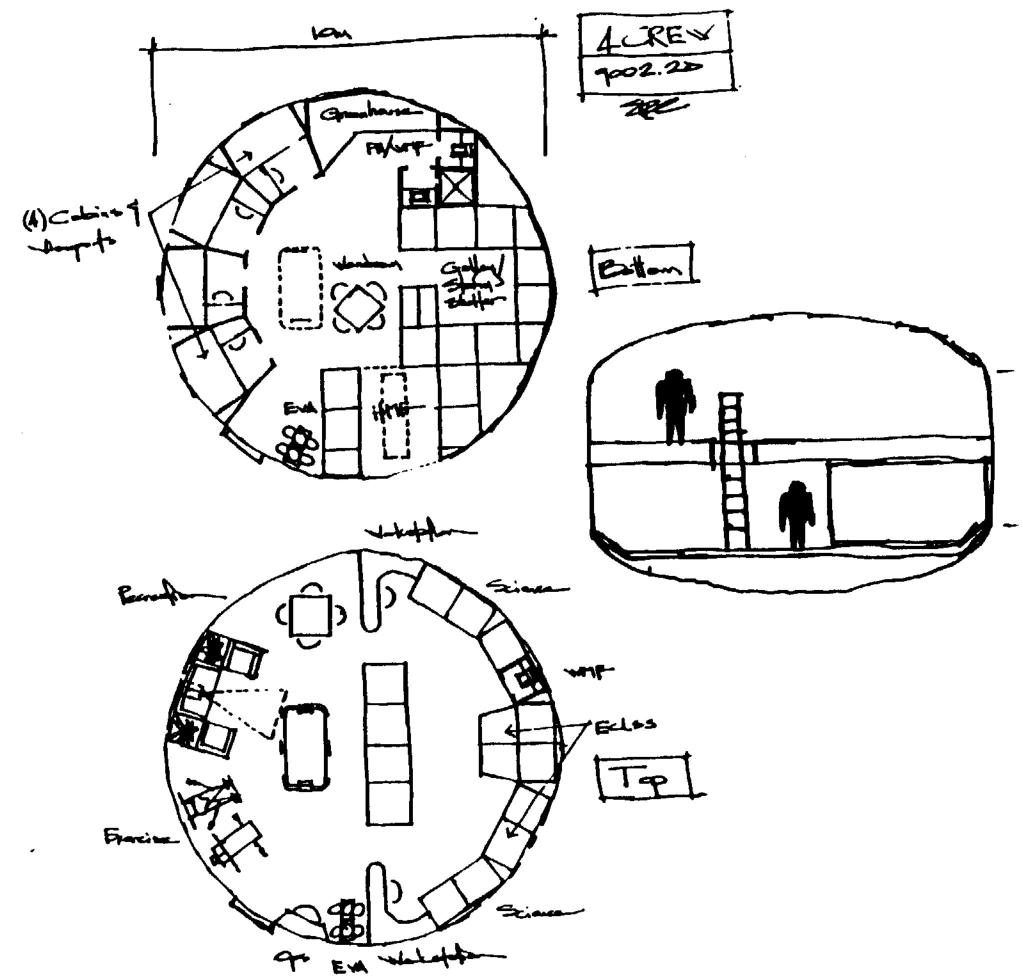
Reference Configurations

The next five charts show preliminary layout sketches of interior configurations developed for crew sizes of 4, 8 and 12, using either simple clusters of small diameter modules or unitary large diameter modules.

Each habitat type has unique advantages and disadvantages from the interior configuration standpoint; however, no "roadblock" considerations were uncovered with these initial studies.

Reference Configuration
4Sg2-2/1 & 4Lg3-h





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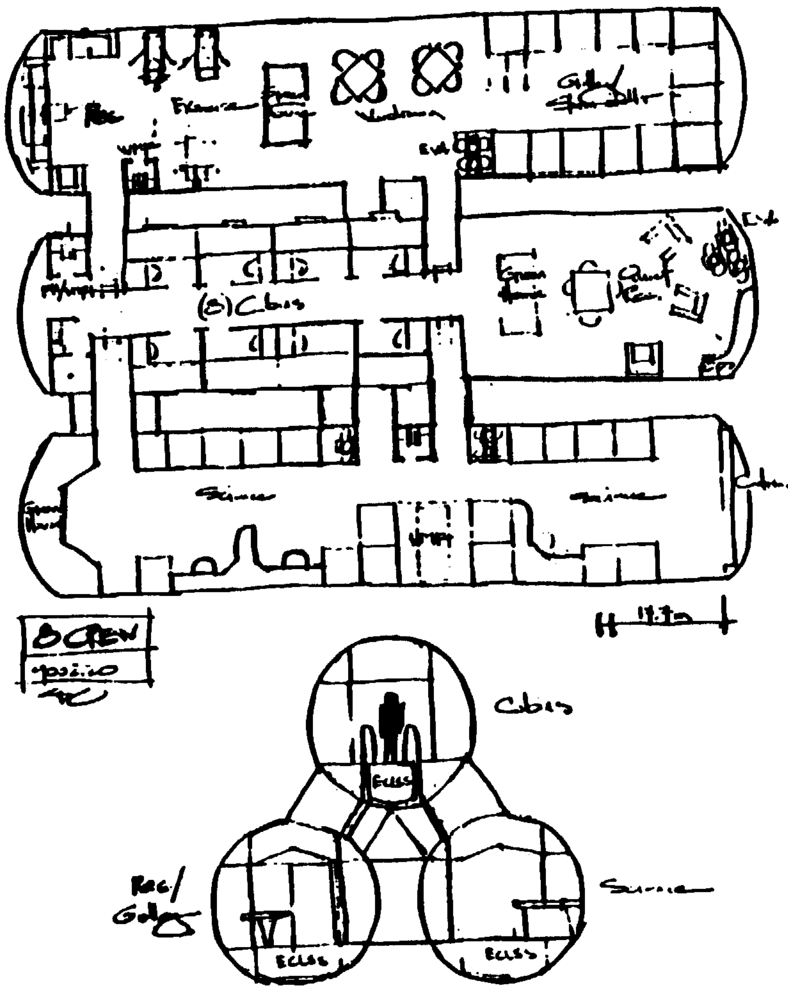
Reference Configuration

8Sg2-3/2



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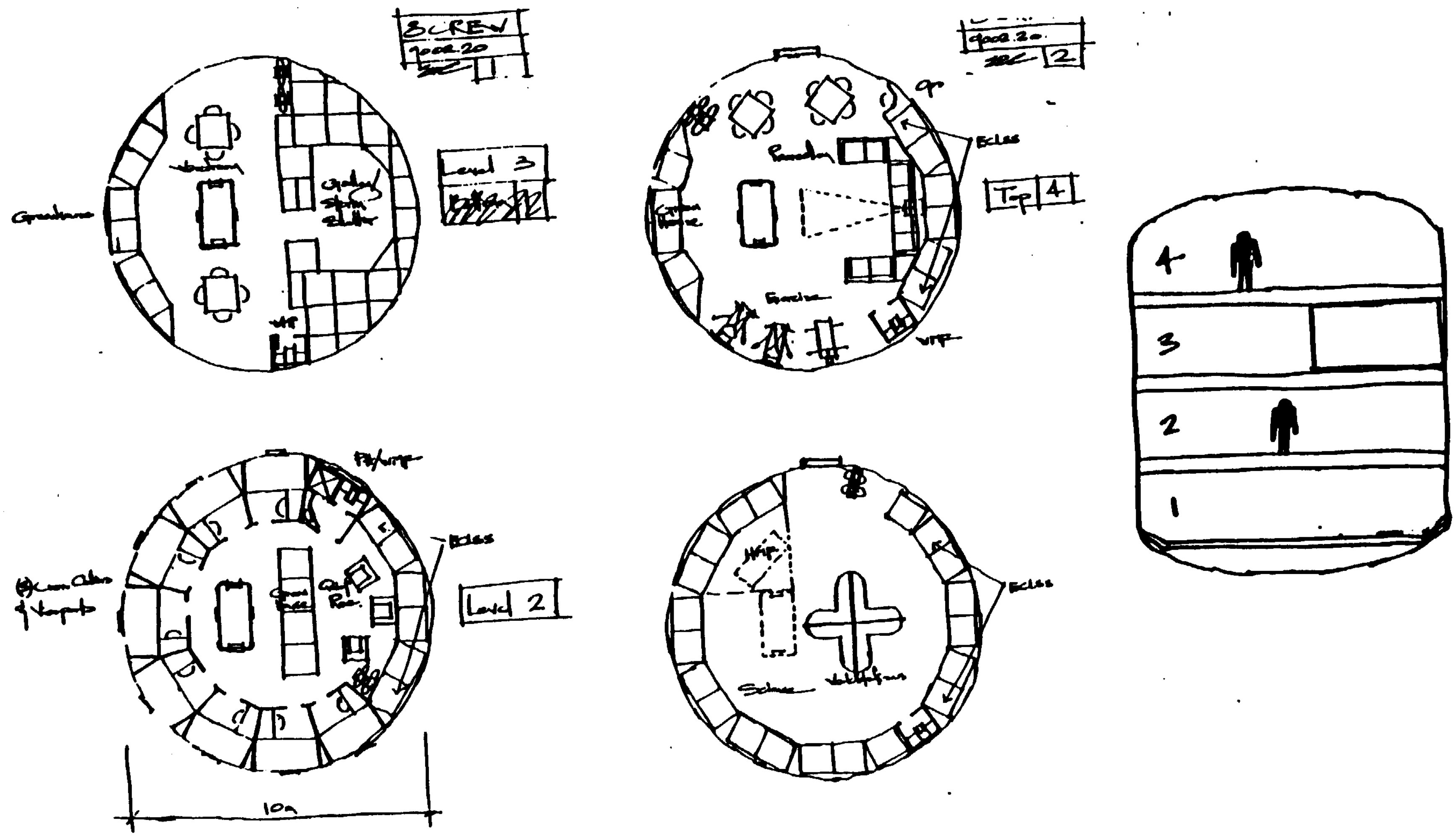
Reference Configuration

8Lg3-h



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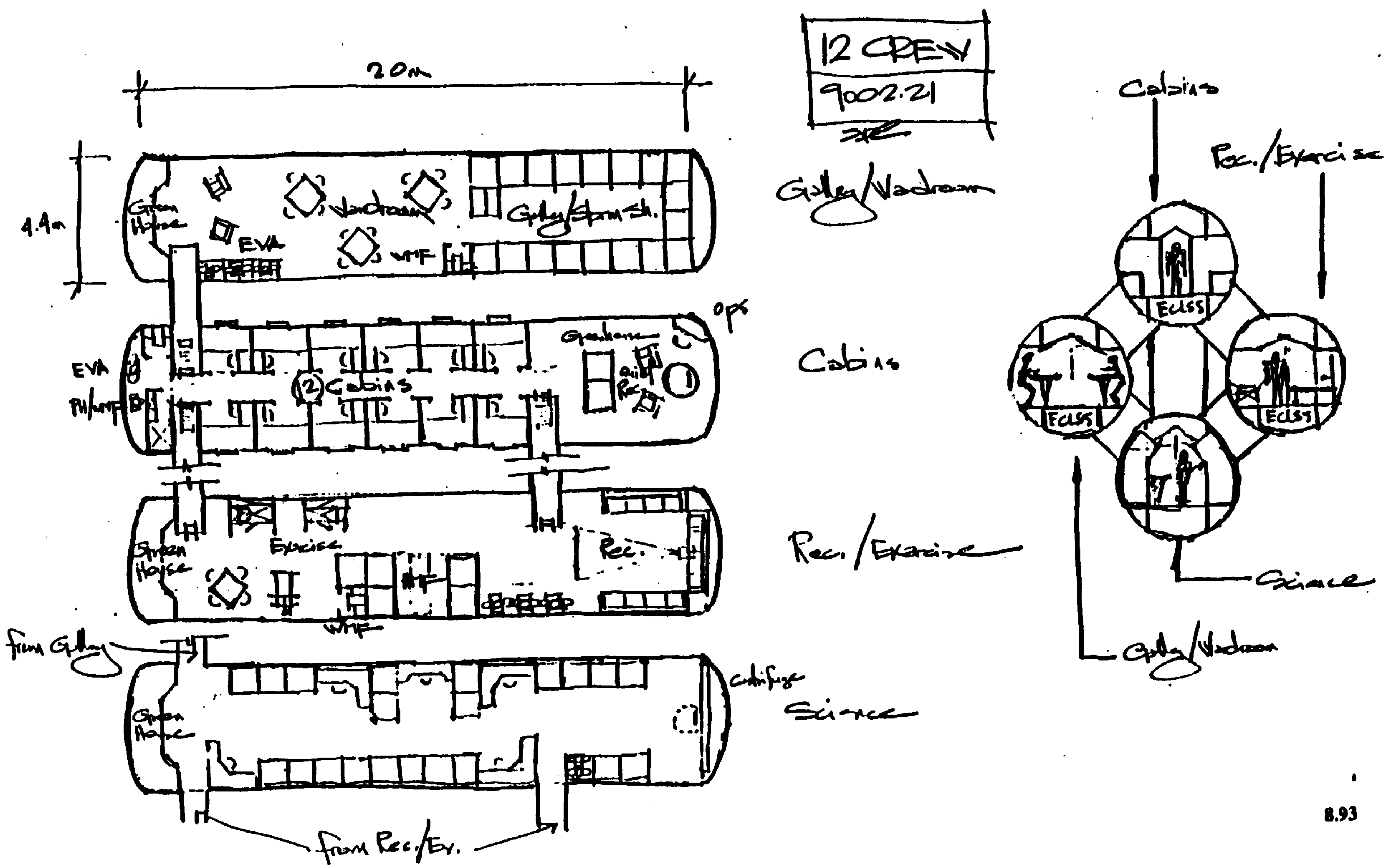
Reference Configuration
12Sg2-4/5



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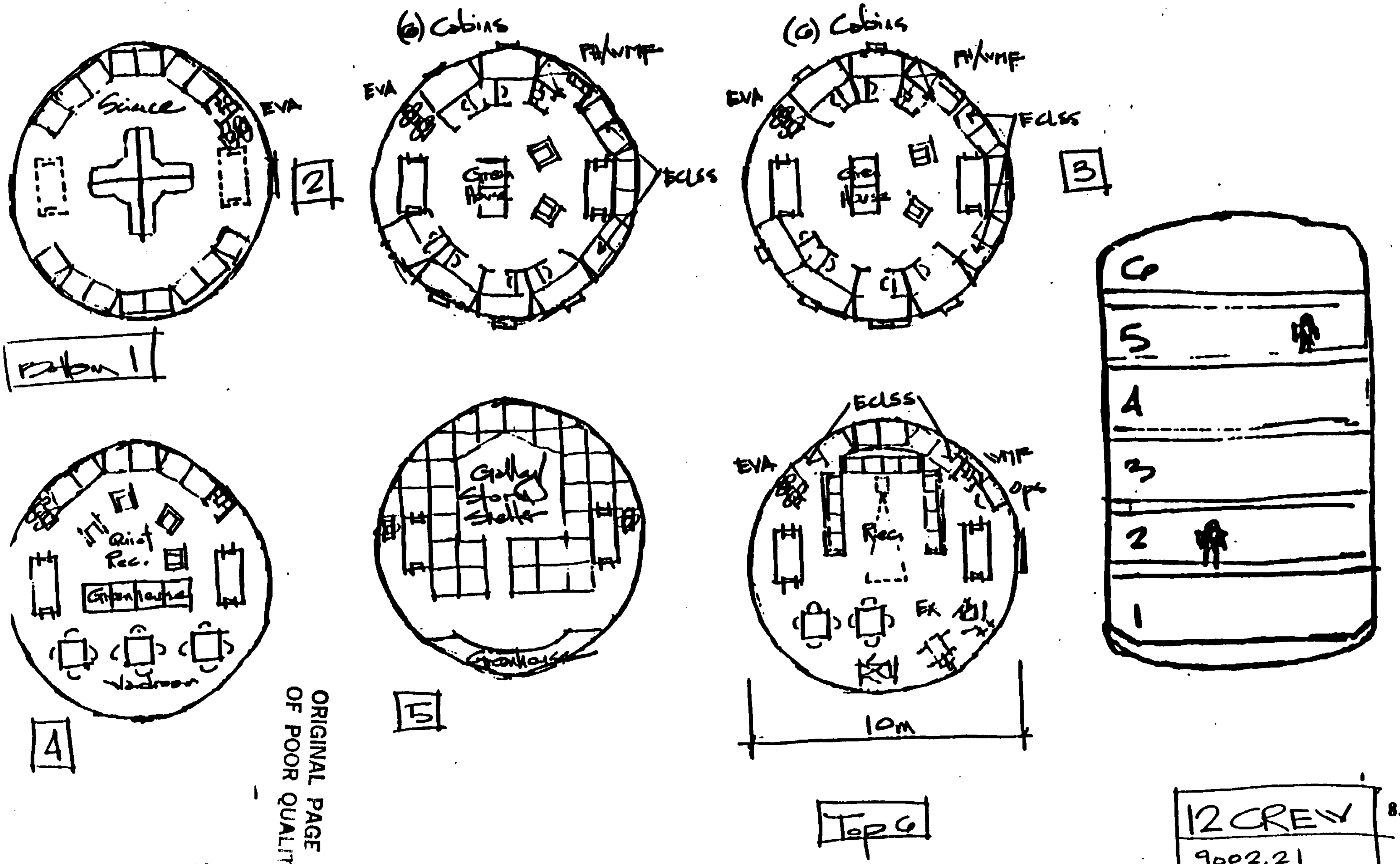
Reference Configuration
12Lg3-h



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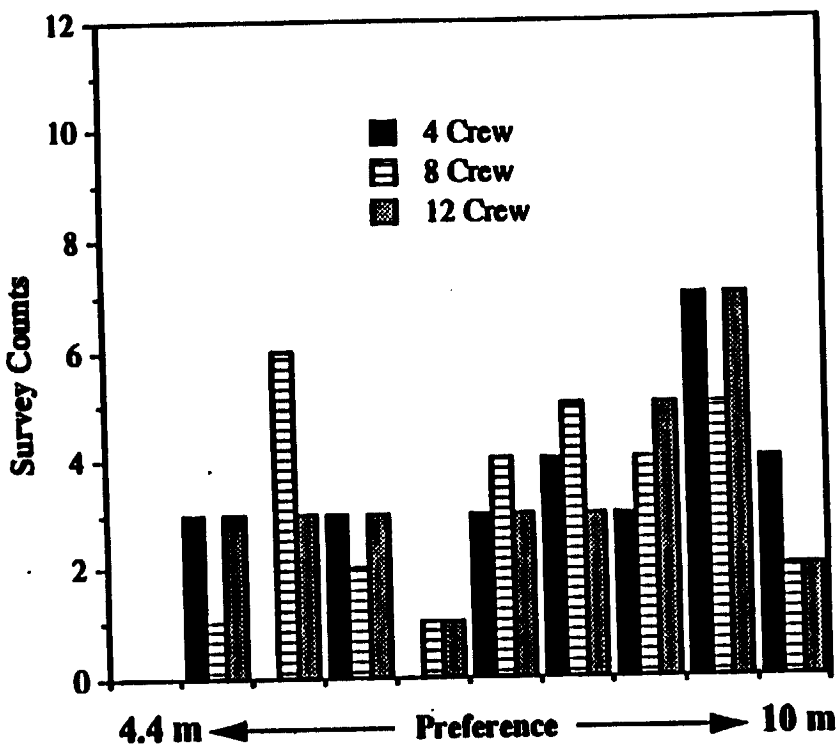
Opinion Survey Results

Using the configuration sketches as points of reference, we solicited the opinions of 56 Boeing employees, asking them to indicate their module type preference on a scale of ten between the small diameter cluster option and the large diameter option, for each of the three crew sizes 4, 8 and 12. We explained to the respondents the type of mission, its maximum possible duration, and the conditions of confinement that characterize it. Each person also had the opportunity to record a simple explanation of the preference indicated. The quantitative results are collected here.

Both classes of respondents showed a statistically bimodal preference. It is not clear whether this is an artifact of the survey technique, or whether people tend actually to develop strong preferences. Engineer-respondents were not as extreme in their preference bifurcation, but tended to prefer the large-diameter option. The breakdown was performed according to engineers and non-engineers because engineers represented the best paradigm available for the type of crew members likely to fly early Mars missions. Many comments were made that the large diameter option seemed more spacious, or "was" larger (even though both options' volumes were strictly the same in all cases). The strongest preference peak for non-engineers was in precisely the place the engineers categorically avoided: complete preference of the small diameter options. A possible explanation, indicated by some of the comments written by non-engineers, is that those people concentrated more on the floor plan than the section cut. In plan, the small diameter option appears more familiar and rectilinear; however, the section (a more accurate experiential estimator of spatial character than the plan, which is a behavioral document) reveals the large diameter option to generate in fact more familiar spaces.

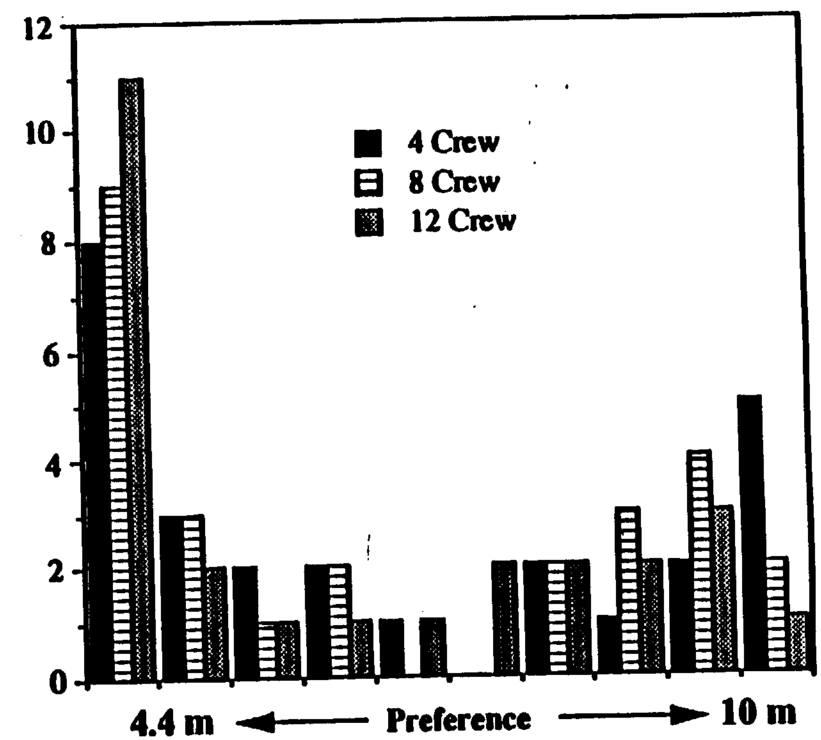
Opinion Survey Results

Responses of 30 Engineers



- Distribution weakly bimodal
- Preference weighted toward large-diameter option
- Sample is probably more appropriate paradigm for technical mission crew

Responses of 26 Non-engineers



- Distribution strongly bimodal
- Strong preference peak for small-diameter cluster option
- Participants tended to focus on floor plan, less attention to vertical section (a more apt indicator of spatial character than plan)

MTV Hab Trade Weight Groundrules

Listed here is the allocation of structural subsystems into discriminators and non-discriminators for the structural mass analysis of the long-duration habitat trade study. Those items which occur in any habitat system regardless of type (EVA-specific and viewing equipment) are not considered. Equipment mounting standoffs are not included because to first order their mass is not expected to be configuration-dependent. Floors and walls, albeit strongly configuration-dependent, were not included because their variations were suspected of being second-order. That turned out in fact to be a valid assumption.

MTV Hab Trade Weight Groundrules

Primary Structure (trade discriminators)

- Pressure vessel
- Structure rings and ribs
- Pressure bulkheads (if any)

Secondary Structure (trade discriminators)

- Inter-module tunnels (if any)
- Inter-module integrating structure (if any)
- Pressure hatches separating redundant volumes
- Meteoroid, debris and thermal protection (surface-area-based)

Secondary Structure (not included; non-discriminators to first order)

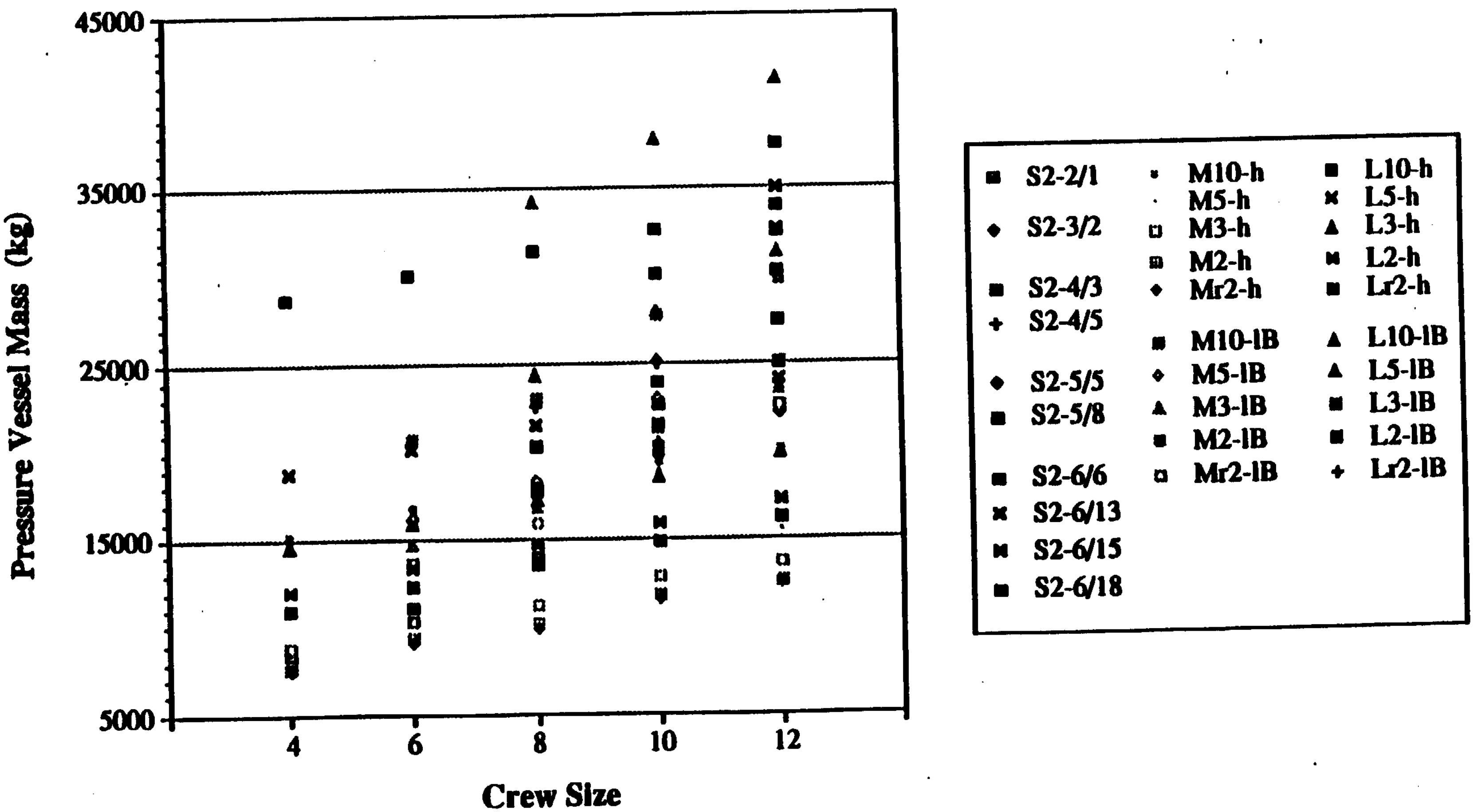
- Airlocks
- Hatches associated with airlocks / EVA
- Windows
- Floors
- Walls
- Subsystem mounting standoffs

Pressure Vessel Mass Analysis

This chart shows the total pressure vessel system masses calculated, including all subsystems just enumerated, for 30 concepts which survived the topology and geometry metric analyses.

Pressure Vessel Mass Analysis

Mass Sensitivity



30 pressure vessel concepts were weighed, covering a wide range of sizes, types and configurations

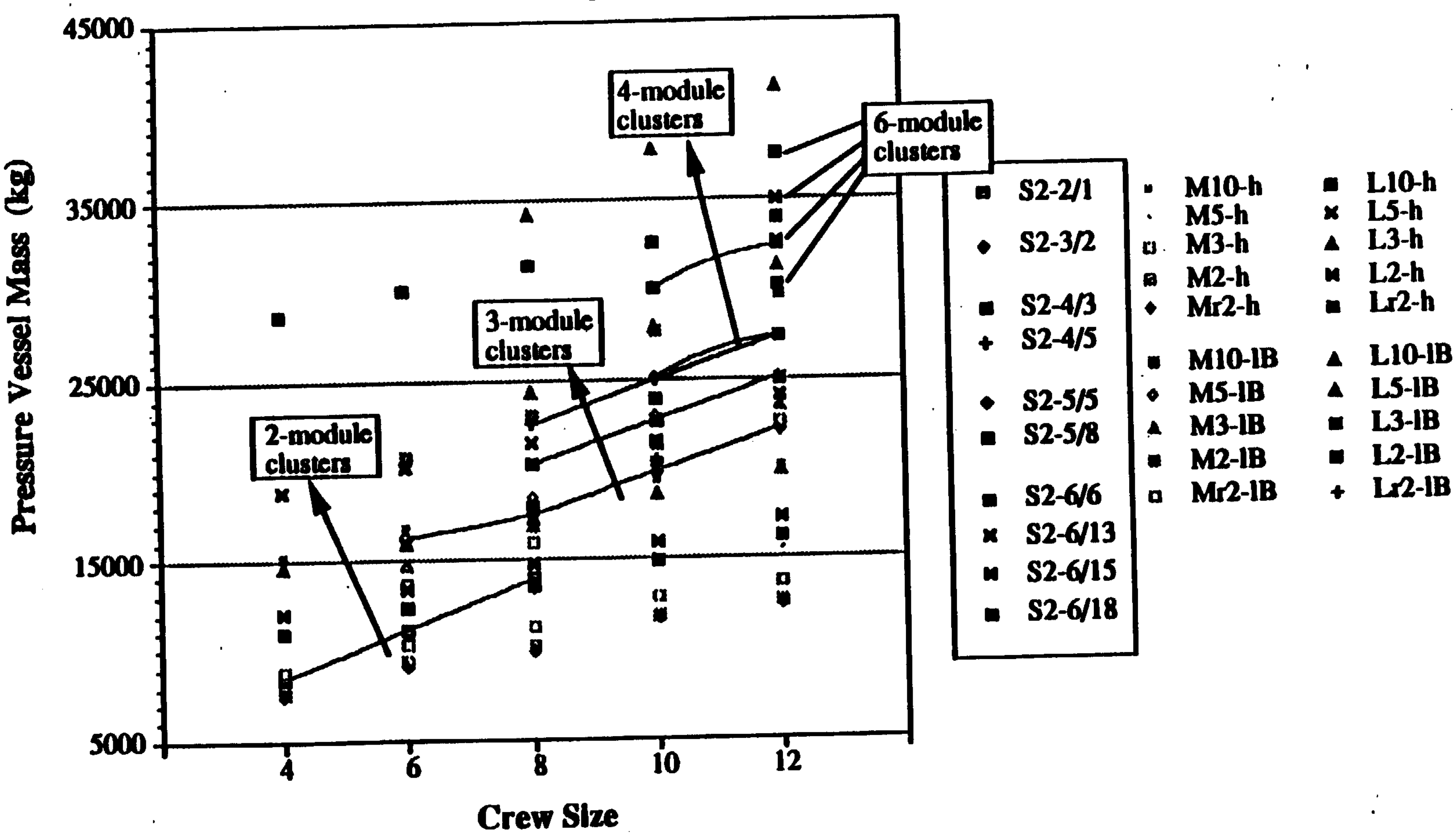
4.4m-diameter Module-cluster Mass Analysis

Plotted as a subset of the 30 concepts are the small-diameter options. Total mass rises rapidly with increasing numbers of modules in the clusters, which penalizes these options for large crew sizes since reasonable single-module length-limits (27 m, commensurate with an HEI Shuttle-C) require clustering for large crew sizes.

4.4m-diameter Module-cluster Mass Analysis

Mass Sensitivity (Small-diameter)

Parameters: Cluster Size
Cluster topology



Clustering modules together weighs much more than extending the modules' lengths

7.6m-diameter Module Mass Analysis

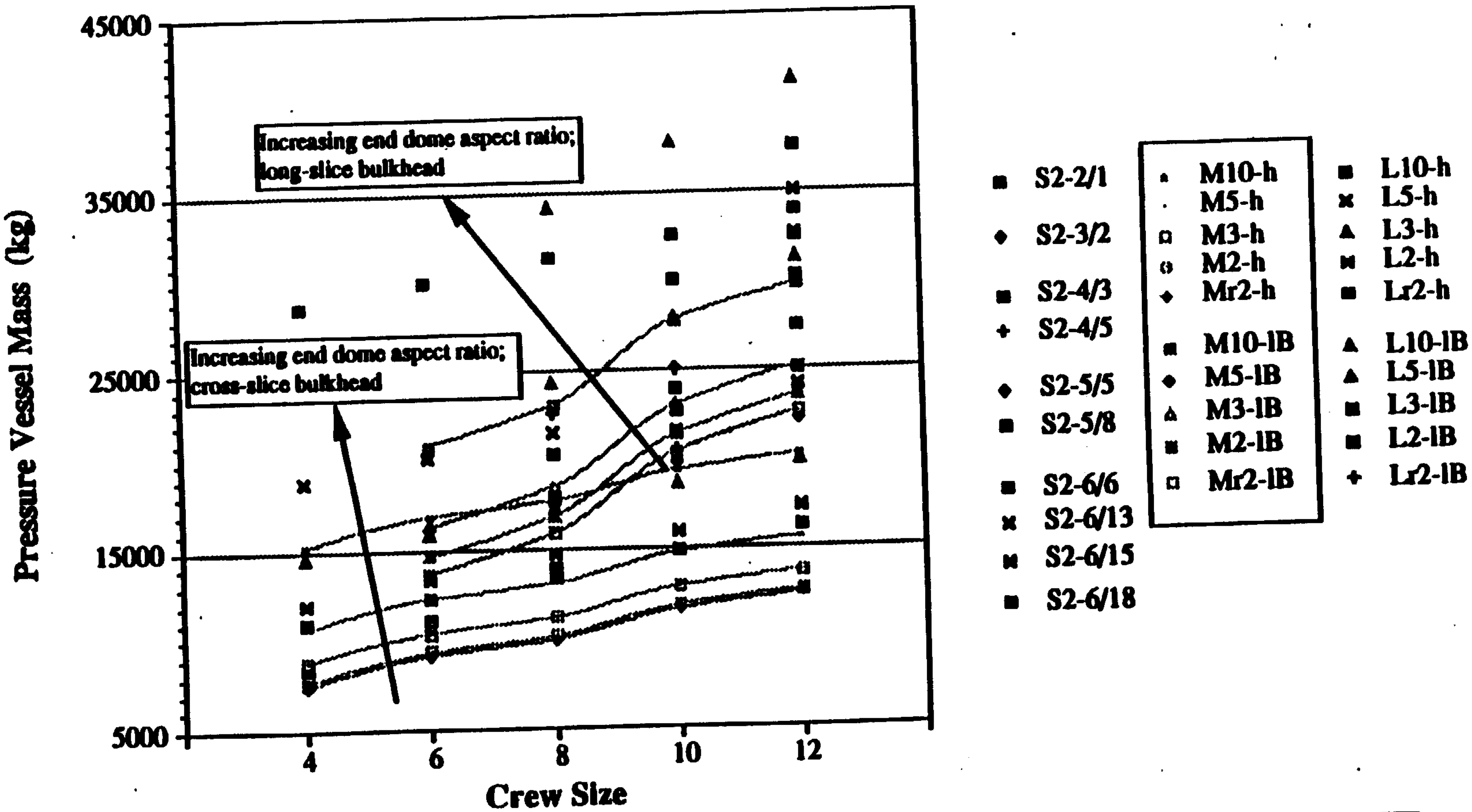
Plotted here are the medium diameter options, both stacked and tunnel-oriented, parametrized according to end dome aspect ratio. Flat end domes are extremely mass-expensive, as is the tunnel orientation (which according to study assumptions has the internal pressure bulkhead running longitudinally).

The curves are not linear because a unit module length increase is achieved by adding 2.8 m (one floor height plus floor structure), which does not correspond precisely to our specific volume assumptions as the crew size increments by one. The module concepts sized for 4 and 8 crew have thus been slightly volume-penalized.

7.6m-diameter Module Mass Analysis

Mass Sensitivity (Medium-diameter)

**Parameters: End Dome Aspect Ratio
Pressure Bulkhead Orientation**



Crew size is a non-linear independent variable for the medium-diameter module; flatter end-domes and longitudinal bulkheads increase mass dramatically

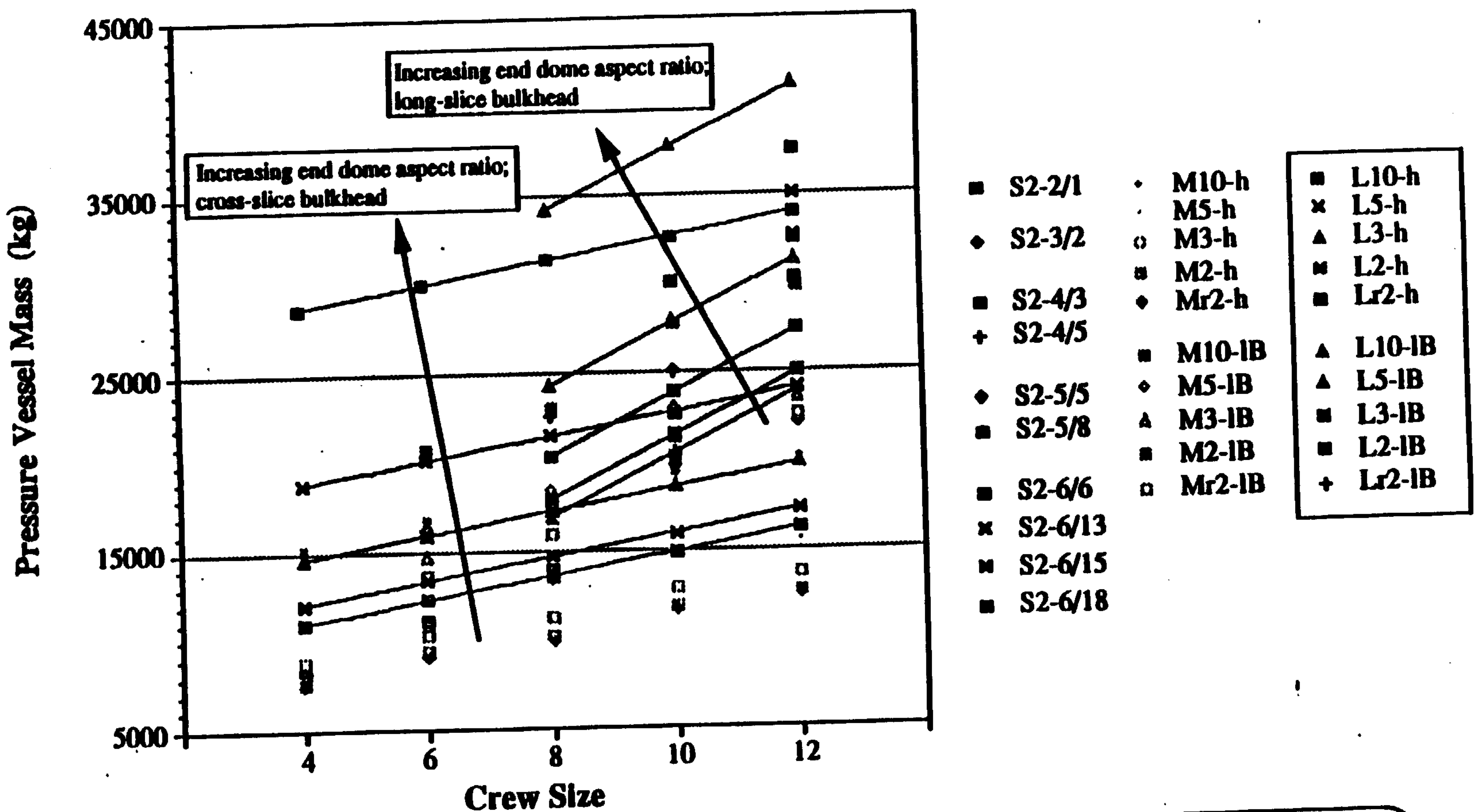
10m-diameter Module Mass Analysis

Plotted here are the large diameter option masses. The plot is analogous to the previous one; however, in this case a one-floor increment in module length corresponds to a one-person increment in crew size, so the curves are linear. The spread caused by various end-dome shapes is more pronounced than for the medium diameter options, because the larger diameter exacerbates the high stresses in the dome shoulder. Dome aspect ratios of 2 and $\sqrt{2}$ are quite close in mass performance.

10m-diameter Module Mass Analysis

Mass Sensitivity (Large-diameter)

Parameters: End Dome Aspect Ratio
Pressure Bulkhead Orientation



Flatter end domes have dramatically increased masses; the longitudinal bulkhead orientation is heavier and is more sensitive to crew size increases

Reference Concept Mass Analysis

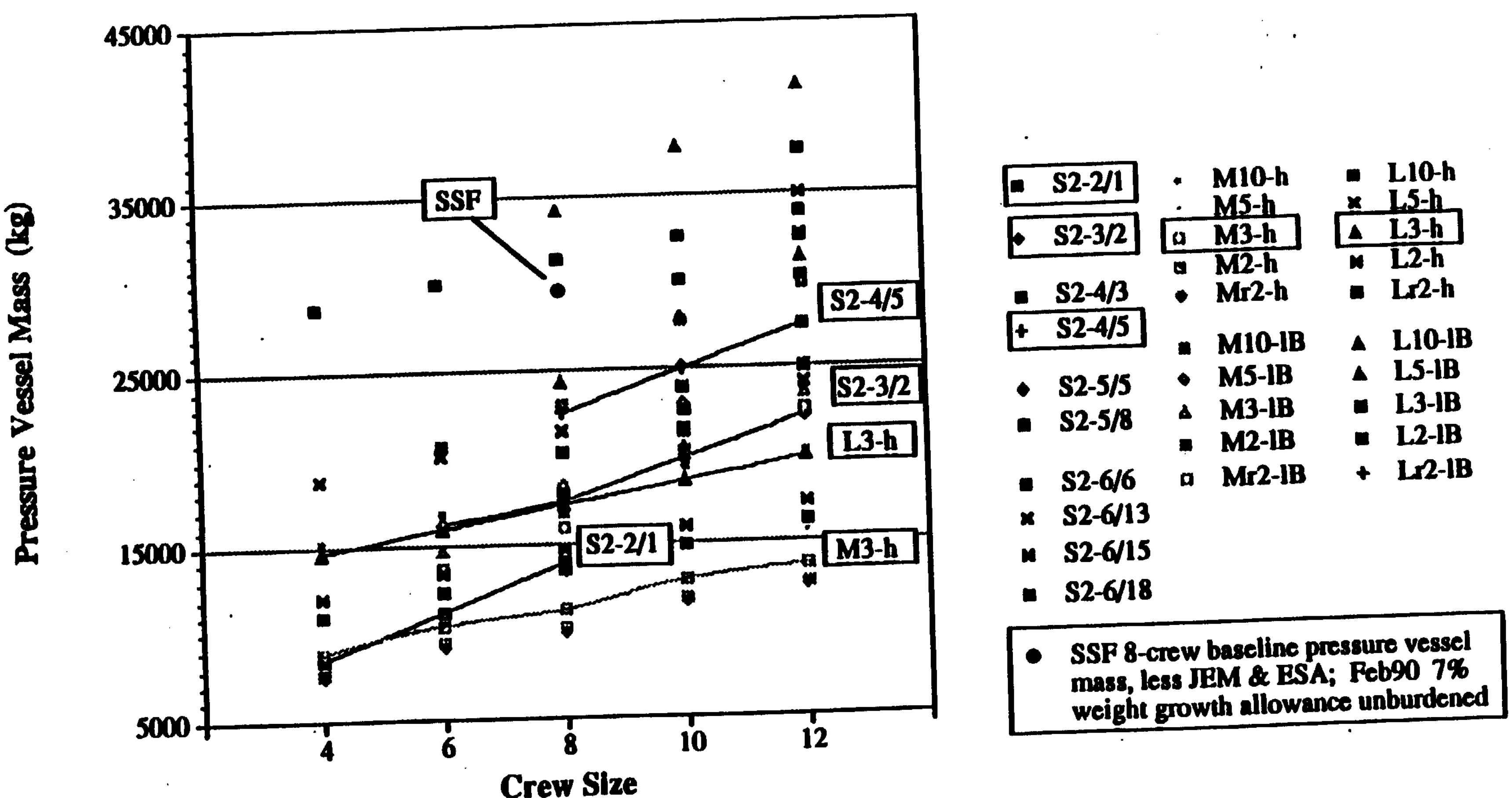
Plotted here is a comparison of reference concepts from all classes of module types. SSF mass is shown for comparison, calculated according to the same assumptions used for the traded module concepts. Although the value does not include the mass of the JEM and ESA modules, it inhabits a quite high weight region; this is primarily due to the heavy end cones of SSF modules, and the particularly mass-expensive topology it baselines (two modules plus four nodes, the rough equivalent of a five-module cluster in our trade study).

The stacked, medium diameter option is seen to trade quite favorably for small crew sizes, and to win handily for larger crew sizes.

Reference Concept Mass Analysis

Mass Sensitivity

Comparison of Reference Concepts



For crew sizes over 6, larger-diameter concepts have an increasing weight advantage over small-diameter cluster concepts

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Outfitting Equipment Mass Estimation

An important consideration for larger unitary module concepts is their ability to be outfitted on the ground prior to launch. Orbital integration is a costly operational burden for an exploration architecture.

The next two charts list assumptions and sources used to develop a parametric outfitting mass estimation algorithm for a Mars-class mission.

Outfitting Equipment Mass Estimation

Nomenclature

F = freezer mass
N = number of crew
E = number of ECLSS strings
M = number of equivalent SSF module volumes
A_p = partition area (m^2).
A_f = floor area (m^2)
P = power level (kW)

Power Levels

Crew	kW
4	25
6	30
8	35
10	45
12	50

Freezer Mass

Crew	kg
4	720
6	938
8	1139
10	1323
12	1491

Equipment	Parametric value (kg)	Comments
ECLSS	1909 * E	Derived from SSF mass, 10 % A&I (attachment & integration penalty)
Sample freezers	50 * N	Derived from SSF mass, 10 % A&I
Food freezers	F	Estimated through preliminary design
DMS/comm & A/V	1560 * M	SSF system mass augmented for long duration mission (LDM).
CHC/exercise	400 * N	SSF system mass augmented for LDM.
Science	667 * N	SSF derived mass: 1/2 of equiv. experimental equip. complement, 10 % A&I
Greenhouse	240 * N	Derived from SSF plant growth facility with 20 % A&I
Wardroom/galley/storage	200 * N	SSF derived mass including ovens, washers, etc, 10 % A&I
Personal hygiene	72 * N	SSF derived mass for shower, handwash, and waste mgt. equip
Storm shelter	2000	Shielding required in addition to configured consumables

(continued)

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Outfitting Equipment Mass Estimation (2)

Equipment	Parametric value (kg)	Comments
Windows	30 + 22.5 * N	2 windows (SSF type) + 1.5 windows/crew member
Crew quarters	250 * N	SSF derived mass, augmented for gravity configuration
Partitions	1.43 * Ap	Double thickness SSF derived partition
Chairs	10 * N * M	One chair per crew member per module
Tables	16.7 * N	Two "table places" per crew member total
Floors	13.3 * Af	*Skylab* -derived Al "waffle grid" floors with beam supports
Finishes & miscellaneous	1 * Af	Floor & wall coverings, hardware allowance for access doors
Power dist. and control sys.	17 * P	Scaled from SSF EPDS
Lighting	73 * M	SSF derived mass
External hatches & bulkheads	694	SSF derived mass

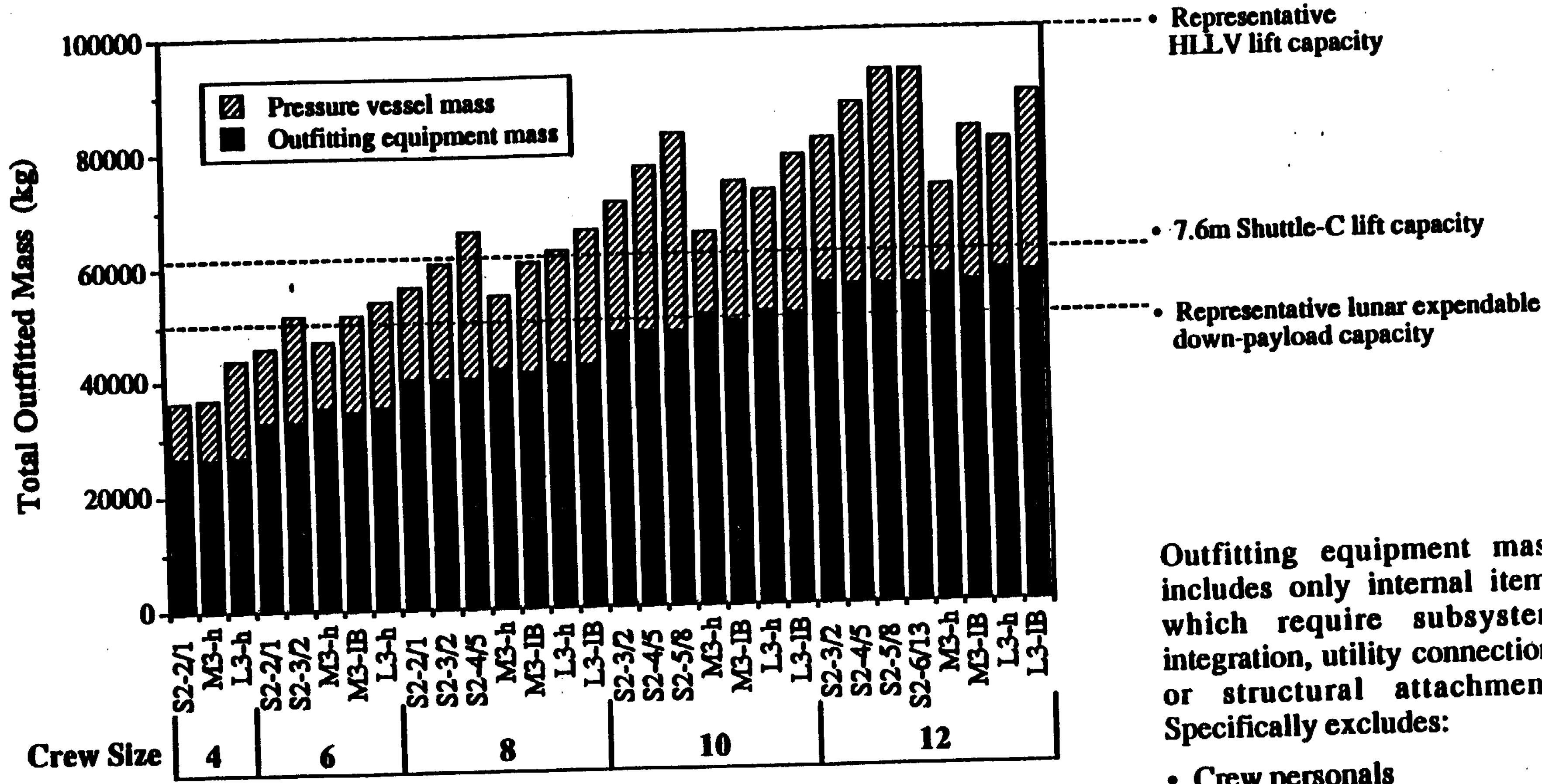
$$M_{equip} = 2724 + (1909)E + (1919)N + F + (1633)M + (1.43)Ap + (14.3)Af + (17)P + (10)NM$$

Long-duration Module Outfitted Mass

Plotted here are the estimated total masses for the 30 habitat concepts brought through the mass analysis. As shown, these numbers do not include items easily integratable on orbit, but only those subsystems which require attachment, connection, test and checkout. The plot shows that within each crew size range, the equipment mass is roughly constant (equipment for stacked module options tends to be slightly heavier than that for tunnel options). Thus, the pressure vessel mass itself is the interesting discriminator.

Given a reference lunar down-cargo capacity in expendable mode of 50 t, we see that some unitary options for crew sizes of 6 can be landed already integrated on the Moon. With the application of detailed weight-reduction efforts, unitary modules for long-duration crews of up to 8 may be accommodated the same way. Clearly, the small-diameter options can be broken up into smaller pieces than their mass totals indicate, for piecemeal launch, landing and integration. Using and ETO launcher like an HEI Shuttle-C in expendable mode, and applying weight reduction efforts to the module concepts, we can see the possibility of launching a module for 10 crew, fully integrated, into orbit. Such a module could be landed on the Moon with some internal systems removed.

Long-duration Module Outfitted Mass



Outfitting equipment mass includes only internal items which require subsystem integration, utility connection, or structural attachment. Specifically excludes:

- Crew personals
- Food
- Water
- ECLSS consumables
- Equipment spares
- External power system

Medium and large-diameter module options for crews of up to 6 can be launched from Earth, and landed on the Moon, already integrated

Habitation Module Fabrication Technologies

This chart lists the essential requirements for module materials, and the prime options available for advanced M&P application to space habitat manufacture.

Habitation Module Fabrication Technologies

Critical Requirements

- Thermal / mechanical stability
- Radiation resistance
- Corrosion and moisture resistance
- High specific strength & stiffness
- Producibility & inspectability
- Damage resistance (toughness)
- Vibration damping capability

Technology Options

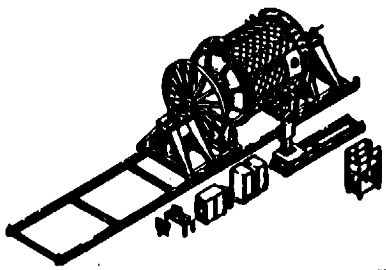
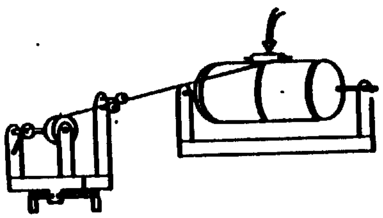
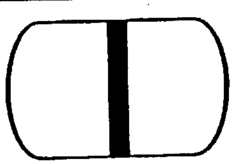
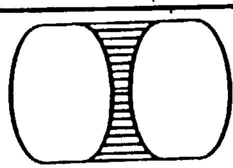
- Conventional welded structure
- Honeycomb core
- Metal matrix composites
- Organic matrix composites

Habitation Module Fabrication Options

Illustrated and elaborated here are the prime candidates for making both pressure hulls, and internal bulkheads for larger diameter modules.

Habitation Module Fabrication Options

STCAEM/jeb/26Feb90

	Options	Suggested Methods of Fabrication
Pressure Vessels	Function: Provide safe habitable volume for crew Assumptions: Near term technologies; external environment exposure; stresses primarily tensile	
	Conventional design - Isogrid - Monocoque	Aluminum Alloy - Welded
	Composite design - Metal matrix - Honeycomb	- Filament wound SiC/ plasma sprayed aluminum, with compaction by hot mandrel - Al face sheet brazed to Al core
Interior Pressure Bulkhead	Function: Provide safe-haven capability in event of hull penetration Assumptions: Near term technologies; internal environment exposure; shear, bending, and tensile stresses	
	Conventional design - Flat panel - Monolithic	Aluminum Alloy - Welded
	Composite design - Concave panel - Honeycomb - Metal matrix face sheet - Organic matrix face sheet	Aluminum Alloy - Brazed or adhesive bonded - Hot pressed SiC / Al, brazed to Al core - Graphite/epoxy layup, adhesive bonded to Al core

Organic Matrix Composites
Metal Matrix Composites

The next two charts survey the features of various fibers and matrices for composite materials that might be considered for habitat construction.

Organic Matrix Composites

Fibers (filament, fabric, or tape):	
Graphite	Causes galvanic corrosion of aluminum. High strength & stiffness, poor vibration damping, low cost
Boron	Very high cost, high compressive strength
Kevlar	Limited compressive strength, good vibration damping, good compatibility with epoxy
Glass/quartz	Low cost, good strength, low modulus, low fatigue resistance, poor adhesion to matrix
Organic Matrix Resins:	
Epoxy	Low offgassing, moderate toughness, thermoset processing, low cost, low temp cure
Polyimide	Potential offgassing, good toughness, thermoplastic or thermoset processing, higher cost than epoxies, high temp cure
PEEK	Higher cost than polyimide & epoxy, thermoplastic processing, high toughness, high temperature strength, repairable by heating
Others	Less suitable

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Metal Matrix Composites

Fibers (filament, fabric, or tape):	
Graphite	Low cost, high strength & stiffness, potential reactivity with matrix alloy
Boron	Very high cost, high compressive strength
SiC	Low cost, compatible with matrix alloys readily processed
Al_2O_3	High cost, potential reactivity with matrix alloy, high temperature stability, lower impact strength than boron
Matrix Alloys:	
Aluminum	Lowest cost, moderate temperature capability, better environmental resistance than Mg.
Magnesium	Moderate cost, combustible, higher temp capability than Al, lower impact resistance than Al.
Beryllium	Very high cost, toxic products, limited supply, favorable thermal properties, low impact resistance
Titanium	High cost, high temperature strength, resistant to corrosion, lower strength:weight ratio than alternatives

Habitation Module Materials Technologies

Summarized and compared here are the synthesized results of our investigation into fabrication technologies and materials options for advanced habitat manufacture. The favored candidate is a composite with SiC-reinforced aluminum matrix. Its combination of desirable structural properties, damage tolerance and environmental inertness indicate the benefit of pursuing technology demonstrations at large scale to generate more data. Its potential for automation could result in cost-efficient production and testing, and its performance advantages over monolithic aluminum would reduce mass and thereby limit transportation costs as well.

ADVANCED CIVIL SPACE SYSTEMS

Habitation Module Materials Technologies

BOEING
STCAEM/jeb/26Feb90

Construction Option	Advantages	Disadvantages
Welded Monolithic Aluminum	Extensive service history Good transverse strength Low cost High damage tolerance Machinability	Highest weight Low specific strength and stiffness
Aluminum Honeycomb Core/ Aluminum Face Sheet	Good in shear & bending Lower weight than monolithic	Low tensile strength High volume penalty Complex design & fab.
Aluminum Honeycomb Core/ Graphite-Epoxy Face Sheet	Good in shear & bending lower weight than Al/Al honeycomb sandwich	Potential corrosion High volume penalty Complex design & fab. low damage tolerance
SiC Reinforced Al Matrix (plasma spray & hot press)	High automation possible Cost comparable to Gr/Ep High strength & stiffness Lower weight than monolithic Tailorable thermal properties Good damage tolerance Good environmental resistance	Limited data Intricate process Technology demonstration at large diameters
Graphite Reinforced Epoxy Matrix	High strength & stiffness Moderate cost Tailorable thermal properties Lowest weight	Sensitive to environment (radiation, temp, etc.) Low damage tolerance Requires metallic vapor barrier

Module Concept Selection

The next two charts record our final, preferred module concept and justify the choice according to the four discriminator categories outlined at the beginning of the trade study. The concept selected for further use in the STCAEM Study is not, *per se*, one of the candidates carried through the trade study, but rather represents a combination of the best features of all the leading candidates.

Several mass-reduction decisions have been incorporated in this new reference concept. First, the medium diameter module was the mass winner. It could clearly fit early HEI ETO launchers, and could either use welded-metal technology or drive more advanced, weight-saving M&P options. The bulkhead is turned the "light" way, crosscut through the module amidships, although the module itself is turned tunnel-oriented for use in gravity fields. The upper floor is located at the module diameter (an average of floor options "B" and "C"), introducing the possibility of using it as a diametral tension tie for further vessel mass reduction (commercial airplanes use this technique). The end dome aspect ratio is 2.

The concept enjoys potential for extensive commonality across exploration architectures, both for spacecraft and surface base applications.

The only perceptual reservation about this concept is that it consistently traded poorly for intrinsic pathway boredom and spatial unit option variety. This means that for long-duration missions, the interior outfitting configuration must compensate carefully, to mitigate perceptions of a severely limited habitable domain.

(continued)

Module Concept Selection

Selection

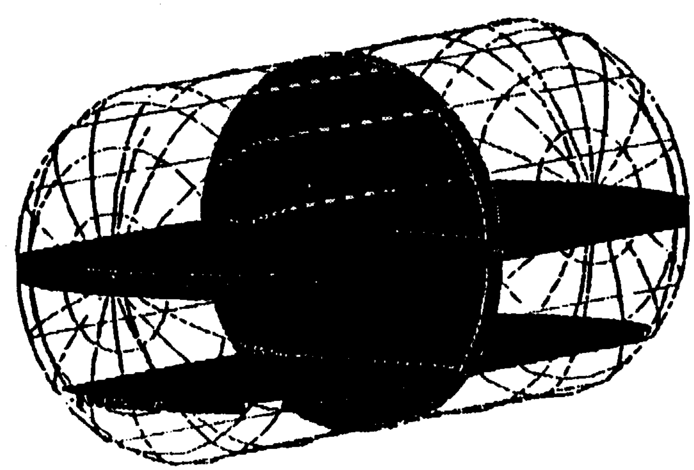
Modified Mg2-l concept family selected for further reference use in the STCAEM study, for

- concept development activities
- trade & sensitivity analyses
- more detailed habitat system definition

Major Features

- 7.6 m diameter
- 2:1 aspect ratio, unpenetrated end domes
- Cross - section, bisecting bulkhead
- Diametral tension - tie, deep floor
- Extensive commonality across architecture: g-field optimized

6 Crew Configuration



Module Concept Selection

The proportions of this module type do not approximate that of SSF modules until crew sizes of about 12 are reached. Beyond that point, it is useful to think of clustering these 7.6m-diameter modules together in simple topologies to extend the habitable domain, for surface bases as well as for large-crew in-space transportation systems.

Finally, it is important to remember that the nature of the trade study has led us to generate a quite conservative habitat concept, which although it combines features demonstrated to be advantageous, still reflects a rather limiting set of assumptions. As a next step, concepts should be considered which combine this reference module type with the smaller diameter module types which we still see as widely applicable throughout all phases of the HEI. For advanced applications, clusters which mix module types and sizes promise good accommodation of functional requirements as well as interesting and stimulating psychological environments.

Module Concept Selection (2)

Functionality

- Unitary vessel minimizes leakage, parts count
- Permits wide variety of internal outfitting designs
- Diametral floor maximizes nominal floor area, facilitates weight-reducing tension tie
- Compact domain, good for access-time safety
- Best overall multi-floor efficiency in g-condition for a range of crew sizes
- Less wall area than smaller diameter; outfitting can compensate

Integration

- Minimizes orbital assembly operations required
- 7.6 m launch shroud likely available for early HEI
- Large crews can be accommdated through simple clustering
- Compact habitat facilitates aerobrake integration

Perception

- Survey results show technical people perceive larger diameter concepts as more spacious
- Barrel vault proportionately invariant with crew (module) size, better than dome
- Module width has better plan aspect ratio than smaller diameters
- Low intrinsic number of unique spatial units; outfitting can compensate
- Lowest score for circulation option boredom over long duration

Cost

- Lightest weight (transportation cost critical for exploration vehicles)
- Welded-metal technology feasible here, well-understood
- Prime opportunity for M&P improvements, however
- End dome complication less than for 10 m size
- Commonality in growth architectures more appropriate for surface system applications

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Evolutionary Lunar and Mars Options

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Agenda

Lunar Requirements
Mini-MEV Requirements
Design for Commonality

LTV/LEV Concept Constraints

Past industry-wide concept development efforts for lunar transportation systems have identified several complicating factors, listed and elaborated here. Successfully accommodating any or all of them severely constrains the configuration options available. The STCAEM Study has adopted all five as design requirements. As they get incorporated into defined vehicle concepts, they will be recorded as design-verified performance requirements.

LTV / LEV Concept Constraints

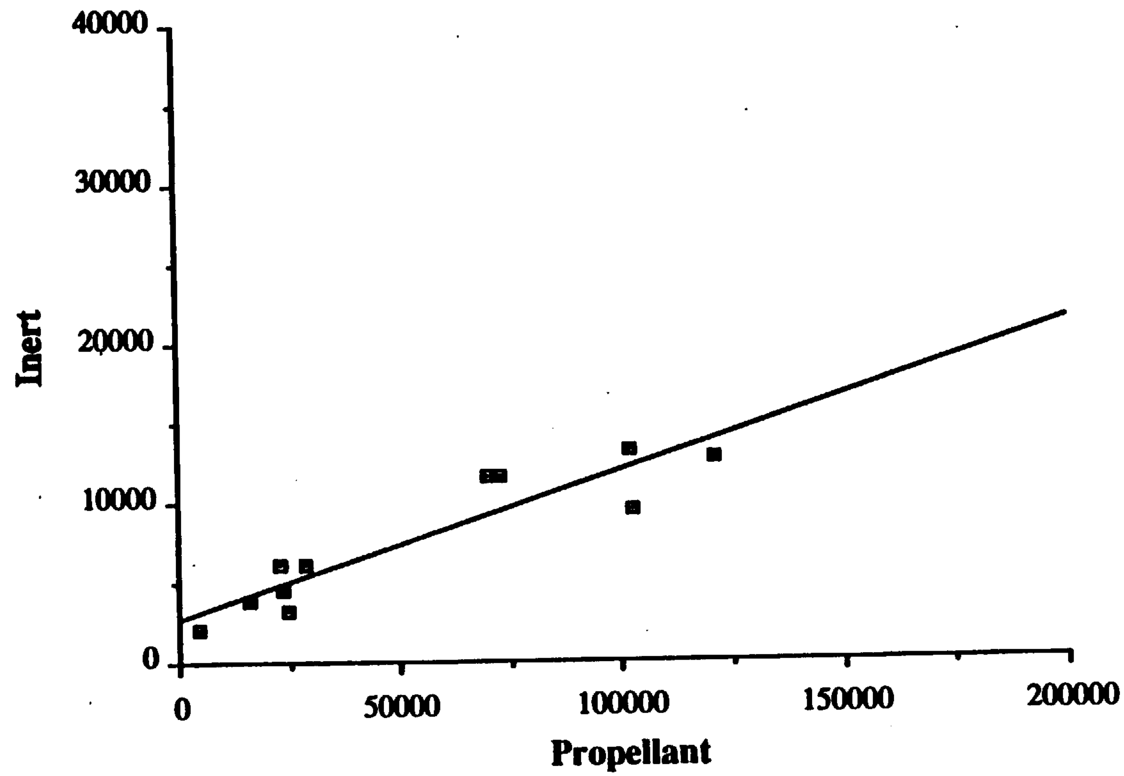
STCAEM concept development directly addresses 5 challenges

- **The geometry of aerobraking** (non-symmetrical relative wind configuration, proper vehicle CM placement, changing mass-balance conditions)
- **Accommodation of mixed payloads** (versatile cargo manifest delivery, transfer between vehicle stages, and processing at SSF)
- **Cryogenic propellant transfer in LLO** (Options: slow-spin pumped; closed thermodynamic with vented chilldown; open-vent quiescent fill)
- **Fully re-usable design** (no drop-tanks; potentially >5 flights per vehicle)
- **Growth capability** (ganged LTVs for evolutionary architectures, higher energy or alternative missions)

Propellant vs. Inert Mass for Lunar/Mars Vehicles

This parametric curve was developed for all of the STCAEM point conceptual design stages to date. A linear regression of these points results in Mass = 2500 + (0.0875)(Propellant Capacity), in kg. This scaling equation was used to develop lunar mode performance parametrics.

Propellant vs. Inert Mass for Lunar/Mars Vehicles



- Data developed from stages designed in STCAEM study
- Linear regression allows parametric sizing
- Mass = 2500 + (0.0875)(Propellant capacity) [kg]

Lunar Modes Performance

Performance of five lunar modes, for crew rotation and resupply missions, is graphed here. Each mode is characterized for IMLEO, LTV stage propellant loading, and resupply (propellant and cargo). The resupply values do not include mass for propellant or cargo carriers. These vehicles are presumed to be fully reusable; no performance distinction is made between tank replacement and propellant transfer refueling.

The direct tandem LTV mode uses a tandem-staged LTV on a direct mode; there is no lunar orbit rendezvous. The relatively massive LTV crew module (weight assumptions shown on the charts are early estimates accommodating a crew size of 6) is taken to the lunar surface with its radiation storm shelter, as is the Earth return aerobrake. This is not an efficient crew mode; its benefit is that no LEV or LEV crew module is needed, and initial development cost is reduced. If crew trips to the Moon are less frequent than once per year, the development cost savings are indicated as economically more significant than the added launch cost. This is an efficient cargo mode (with the lunar lander LTV left on the lunar surface, until such time as surface refueling with LLOX becomes feasible). A vehicle sized for crew rotation and resupply (based on an integral number of HEI-Shuttle-Z ETO flights for the resupply) can land about 50 t on the Moon.

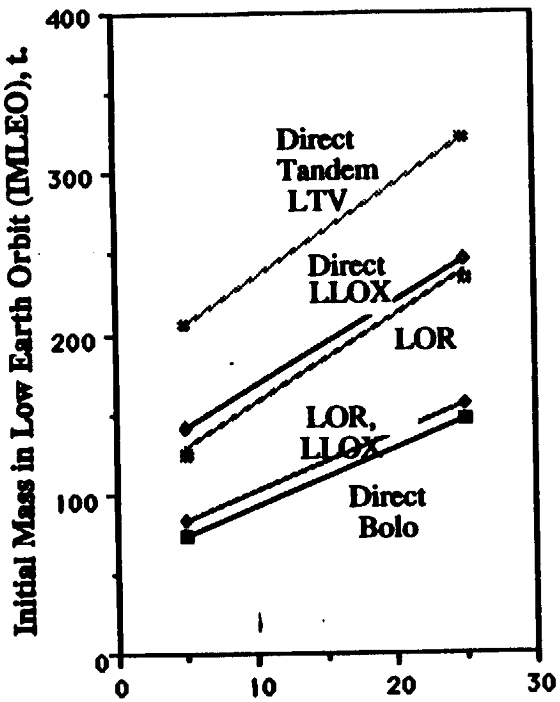
The direct LLOX mode uses lunar oxygen for the return trip. LOR is the conventional LOR mode, with an LEV crew cab of 3500 kg. The LOR/LLOX mode uses lunar oxygen in the LTV, and it is presumed for the data presented here that the LEV returns to lunar orbit with enough oxygen for its next descent. If lunar trips are infrequent, this may not be practical. In that case, the LOR/LLOX mode can use lunar oxygen only for LEV ascent; its performance is about the same as the direct LLOX mode.

The direct bolo mode uses a rotating tether (bolo) in lunar orbit, as outlined on the following page. The mode is very efficient, but the mass of the lunar orbit bolo is large, and several years' lunar transfer operations may be required to emplace it. From a mass and operations standpoint, then, the bolo mode appears to provide a costly and only marginal performance benefit over the use of LLOX.

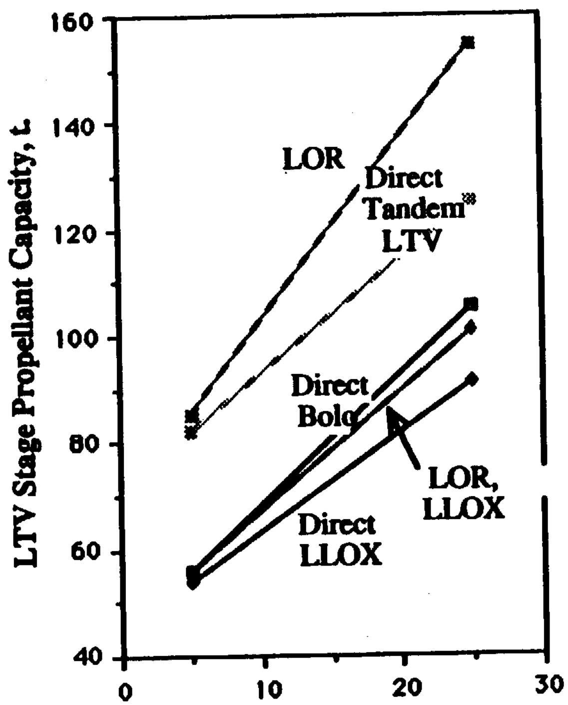
Lunar Modes Performance

Crew Mission, 1 t. Payload Returned

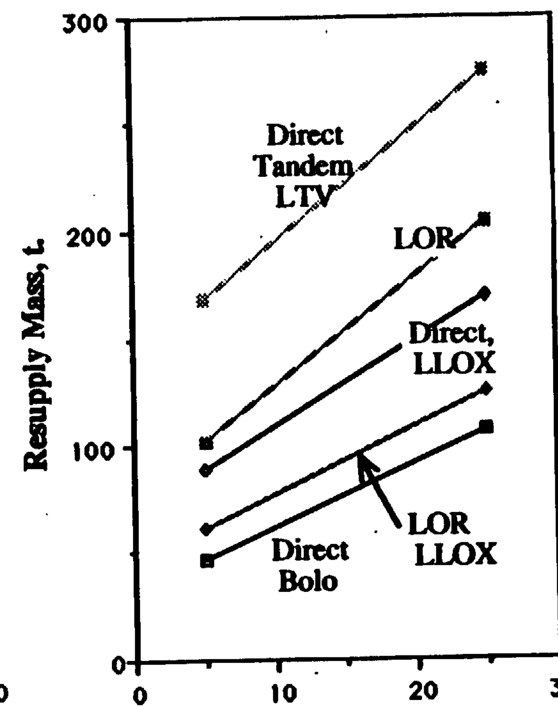
IMLEO Performance



Propellant Loading



Resupply Performance



Note : LTV crew cab mass 8.05 t; LEV crew cab mass 3.5 t.
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ADVANCED CIVIL SPACE SYSTEMS

Lunar Modes Performance
Cargo Delivery Mission

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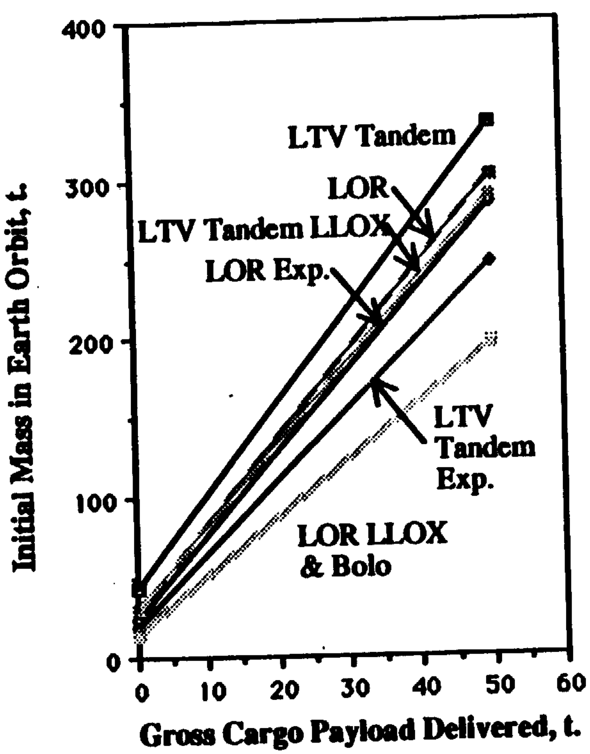
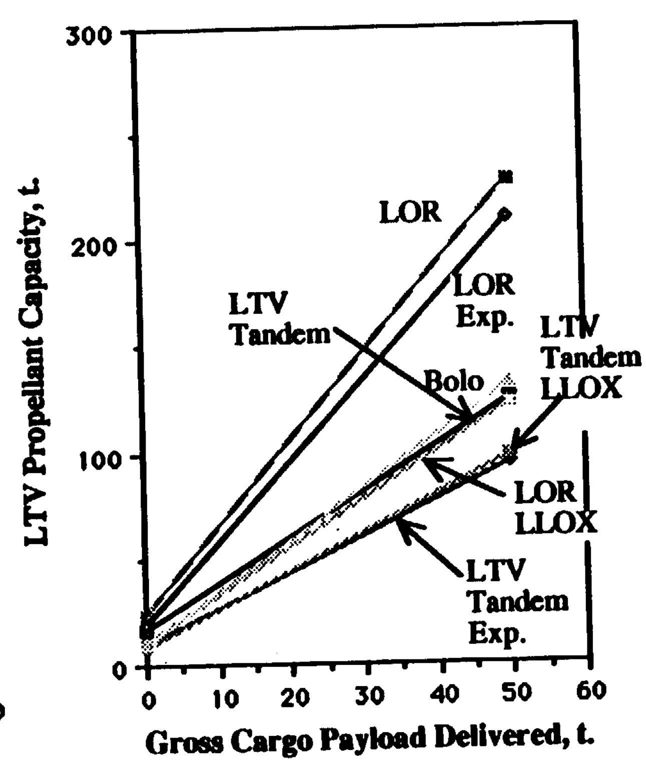
STCAEM/grw/20Mar90

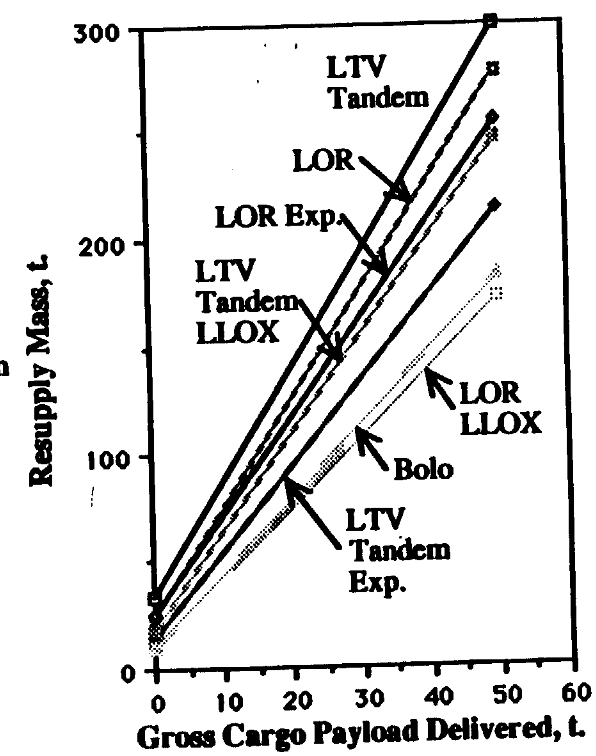
IMLEO Performance



LTV Capacity



Resupply Performance



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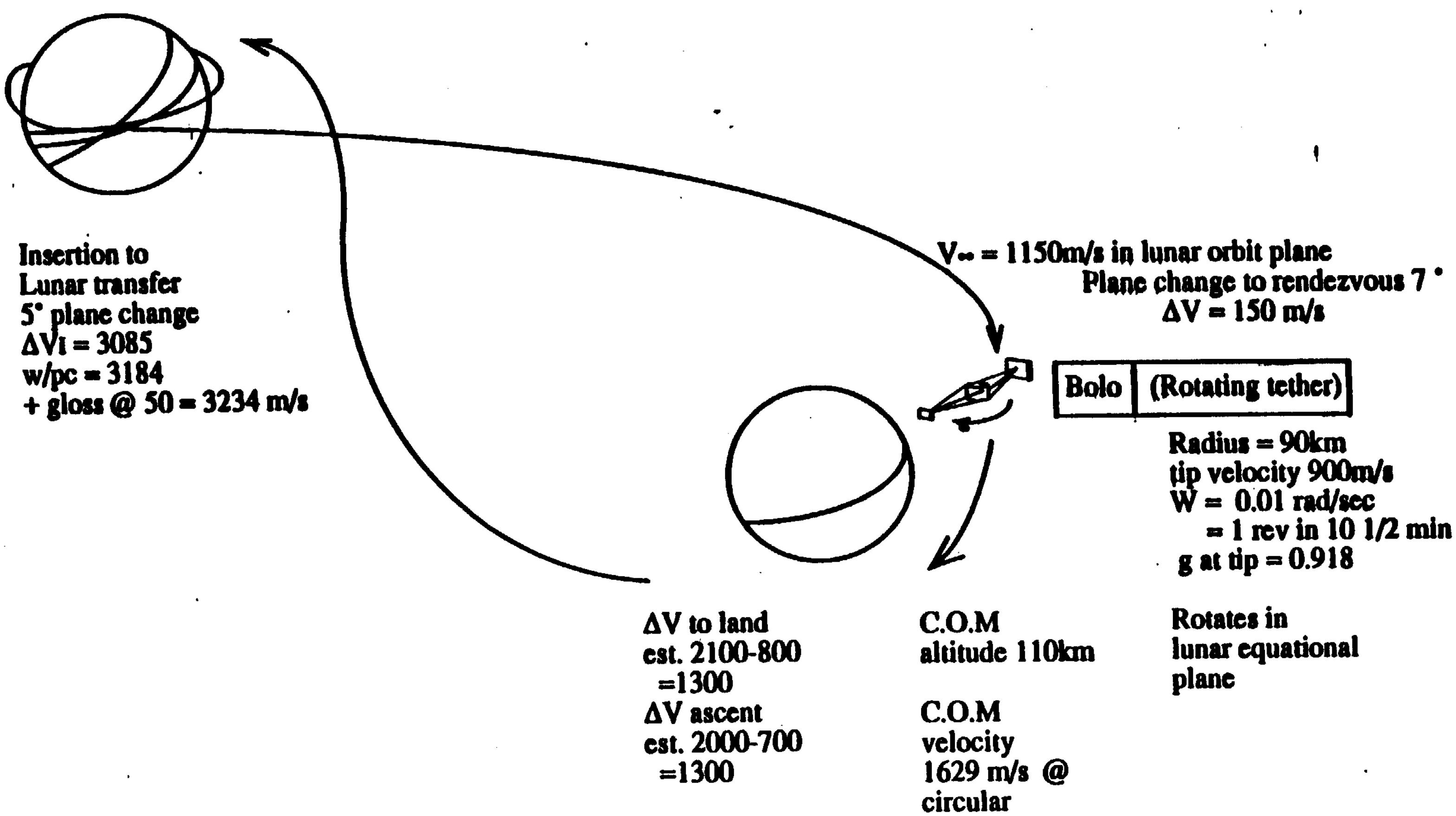
Lunar Orbit Bolo Concept



Mini-MEV Concept

A cis-lunar transportation system sized according to the preceding discussion is of the same scale, and thus allows the potential for commonality, with a down-scaled Mars lander concept, called here the "mini-MEV".

The mini-MEV was conceived as a trade alternative to the reference MEV concept, for the reasons cited on this chart. From a performance-modeling standpoint, the concept warranted concept definition, and was pursued in parallel with the evolutionary lunar system concept just outlined.

Mini-MEV Concept

Question:

- Reference MEV is ~ 82 t
- Is it possible to develop a 41 t "Mini-MEV" and perform two landings for the price of one?

Results:

- 41 t MEV, crew size of 3
- Optimized for sortie missions, not base buildup cargo delivery
- 1 t surface payload including rover and small science
- Nominal mission sequence allows multiple landing sites
- Autonomous landing allows surface rescue by second MEV
- Early, scaled-down Mars mission also feasible (e.g 2010 "good" opposition opportunity
- Potential commonality with LEV

Space Transfer Design for Commonality

This matrix summarizes the required design features, in several subsystem categories, for an evolutionary LTV/LEV system, the mini-MEV, and a small MTV to match which could be applied to achieve the 2010, "easy" opposition opportunity. The right column collects similar requirements into a set of design parameters which would encourage direct commonality between lunar and early Mars missions, based on cryogenic propulsion and aerobraking technology. Aerobrake structures tend not to scale well geometrically, quite apart from the requirement to tailor the structural weight of each to its design payload (so that its mass-reducing benefits can be realized); however, for the purpose of this investigation we have chosen an identical size to work with for both Earth return from the Moon and Mars landing.

Space Transfer Design for Commonality

	LTV	LEV	Mini MEV	2010 MTV	Design Case
Crew cab	- 6 crew, 9 d - rad shelter - partial closure	- 6 crew, 3 d - contam. ctrl. - open ECLSS	3 crew, 10 d - contam. ctrl. - open ECLSS	- 3 crew, 1020 d - rad shelter - closed ECLSS	- 4.4m diameter system - modular lengths - optional rad shelter
Avionics	- Deep space - Orbital - R & D	- Terrain - Orbital - Rend & Dock	- Aerobraking - Terrain - Orbital - Rend & Dock	- Deep space - Aerobraking - Orbital - Rend & Dock	- Modular avionics
Propellant tanks	110t load, cryo	25t load, cryo	- 21t load, cryo - vacuum jackets	140t load, cryo	- 25t tankset, vacuum jacket upgrade - 110t tankset
Engines	3 30klbf 1 engine-out	3 30klbf 1 engine-out	3 30klbf 2 engine-out asc.	3 30klbf 1 engine-out	- 30klbf engine - structure & plumbing
Structure	- strut frame - 7.6m pieces integr. on orbit	- strut frame - 7.6m compat. launched intact	- strut frame - 7.6m compat. launched intact	- strut frame - 7.6m pieces integr. on orbit	- common approach
Landing legs	--------------	22m footprint	16m footprint	--------------	- common approach
Aerobrake	- L/D = 0.2 - 23m length - engine port	-------------	- L/D = 0.5 - 26m length - engine port	L/D = 0.5	- 0.5 L/D shape - 26m length - opt. engine section
Payload accommod.	- mission unique - transferable	- mission unique - transferable	- 1t total - rover & science	- comsats - transit science	- lunar system pallet - Mars unique attach.

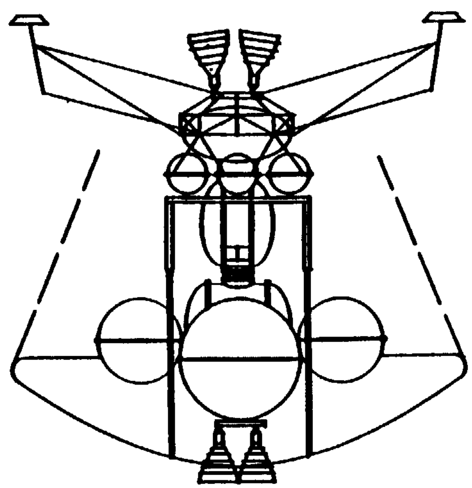
LTV/LEV Configuration

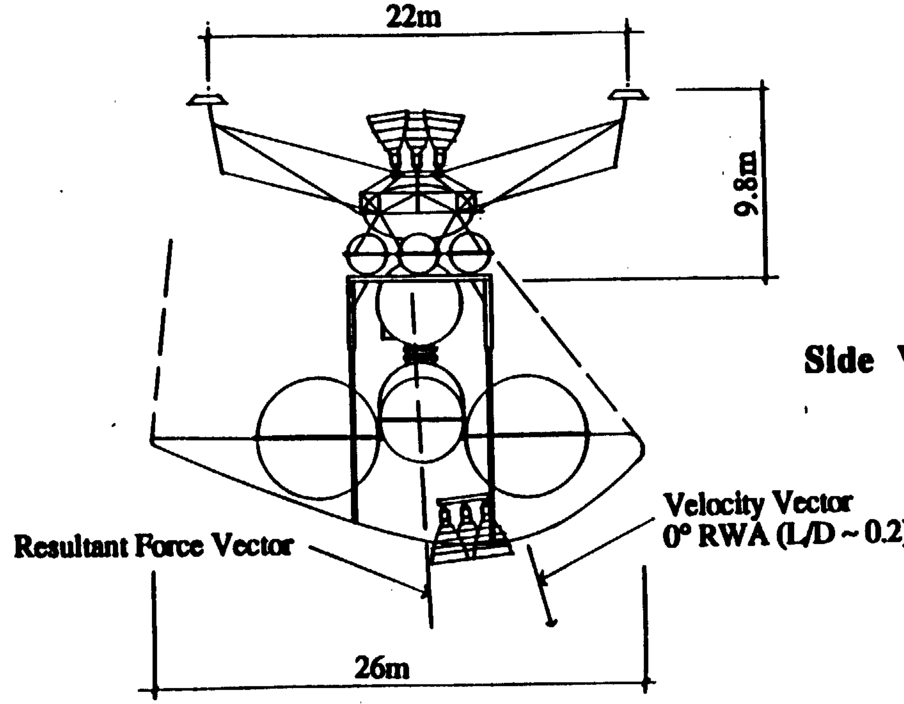
This and the next chart sketch configuration concepts for the cis-lunar system which uses the common parameters just developed. Sized for the Mars case, the aerobrake is somewhat larger than strictly required by the lunar mission, despite retaining all propellant tanks throughout the mission profile. Furthermore, it has the higher-L/D shape required for Mars, albeit flown here in a lower L/D attitude. The LTV engines, while oriented to accommodate the vehicle's changing mass center, are positioned according to Mars landing requirements. Direct transfer of crew from LTV to LEV is accomplished in the same configuration as propellant transfer (whether pumped with rotational settling or transferred using a μg technique).

A single, unconstrained payload pallet is transferred at this time also. The pallet can be integrated at SSF separately from the LTV processing, then mounted for TLI, transferred to the LEV, and unloaded on the surface by a straddling payload transporter. The LEV's height is reduced as much as possible, given the constraint of engine-out on ascent, to maximize unloading efficiency. The landing gear would permit settling the LEV lower to the ground after touchdown to facilitate unloading as well, and are configured in plan to accommodate a triangular straddler. The center section of the pallet is removable, and passes over the LEV-mounted crew module for cargo transfer during crew missions (heavy, bulky, singular payloads like habitat modules or process reactors cannot be accommodated on crew flights due to mass capacity considerations anyway), allowing manifesting of resupply cargo. For unmanned cargo flights, the full payload pallet would be used. The pallet retracts close to the LTV tanks for the aeromaneuver upon return to Earth.

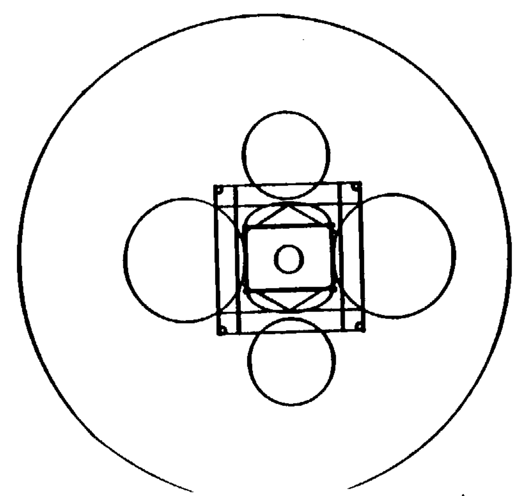
LTV/LEV Configuration

Front View





Side View



Top View (LTV)

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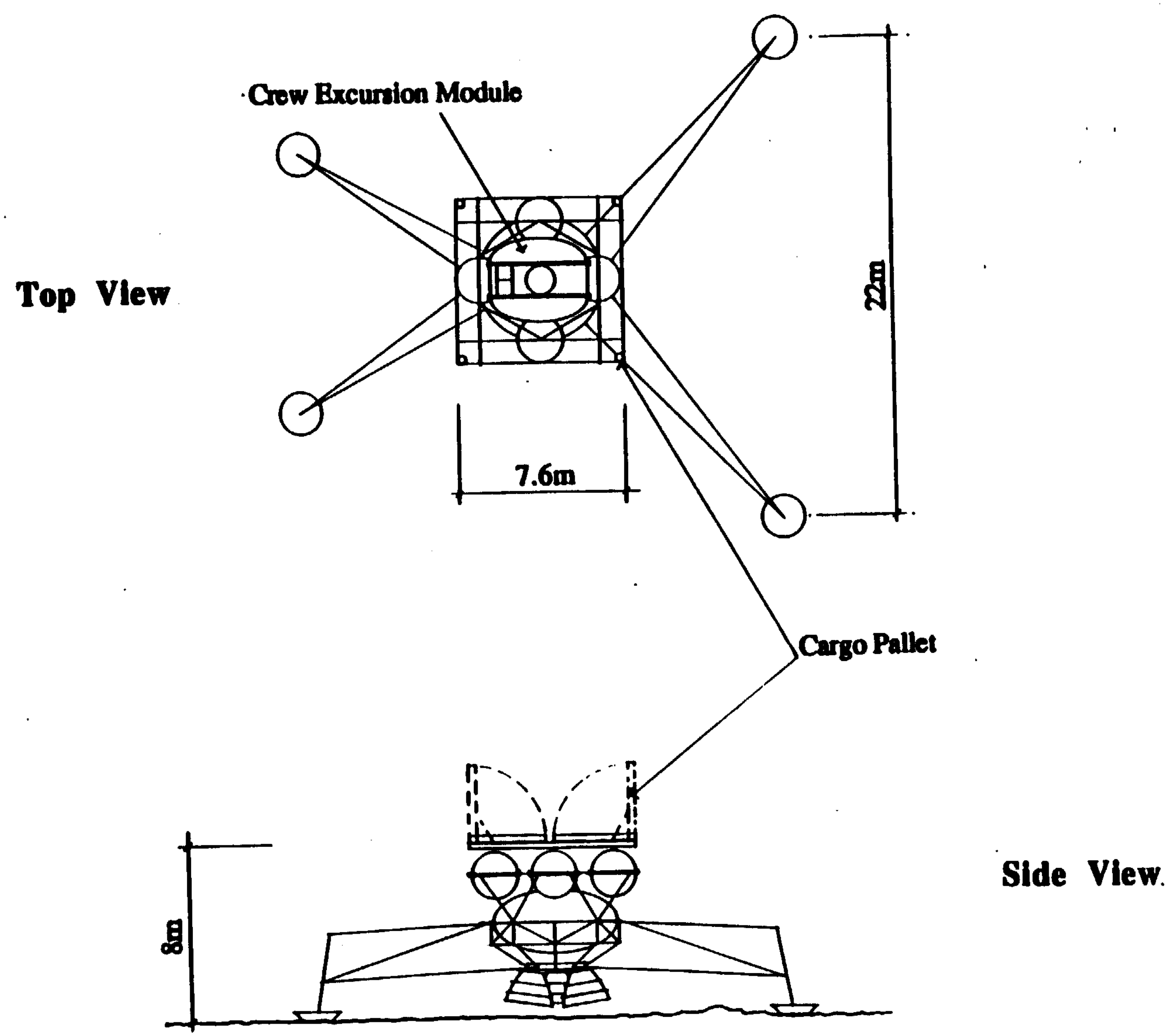
LEV Surface Configuration

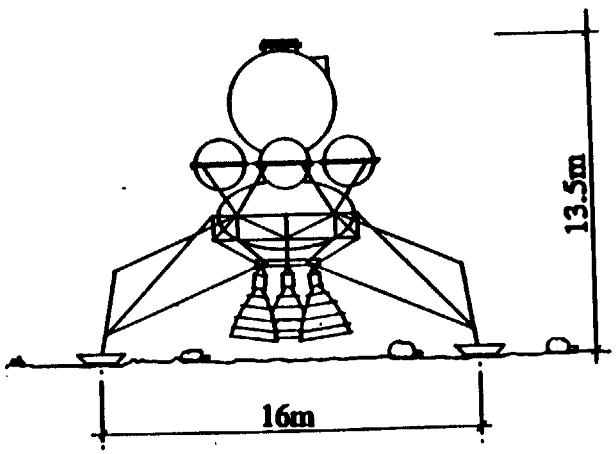


Mini-MEV

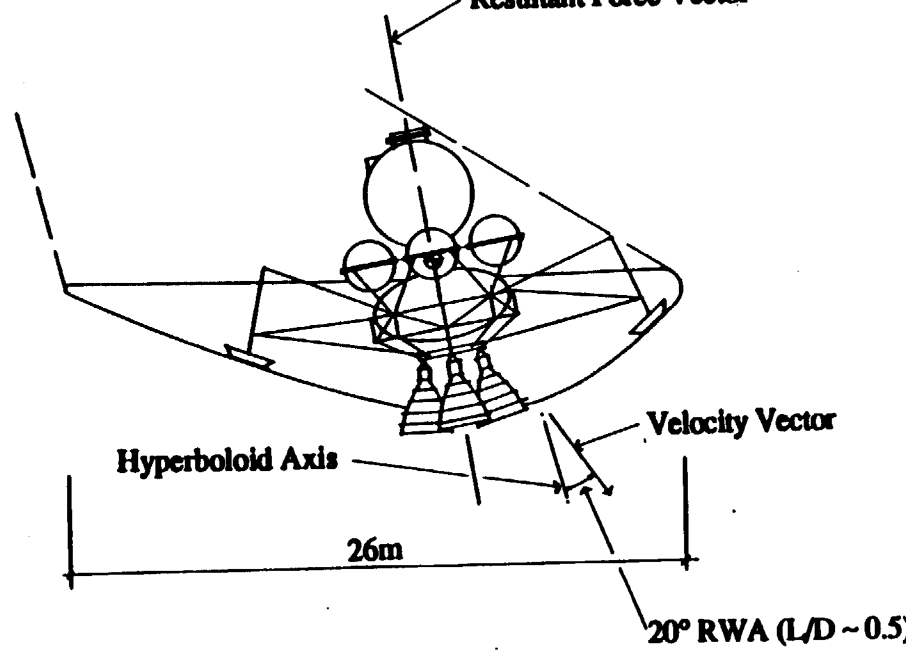
Shown here is a configuration sketch for the mini-MEV, designed according to the same parameters as the evolutionary lunar system just described. The aerobrake is as small as it can be, given the already-minimized MEV height and a requirement for $L/D = 0.5$.

Commonality, exercized through the maturation of a system into meeting the performance requirements for a later mission, has great potential to keep program costs down. It requires pulling a reference concept toward distinct performance goals to develop approaches capable of satisfying both.

Mini-MEV



Surface Configuration



Aerobrake Configuration

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NTR

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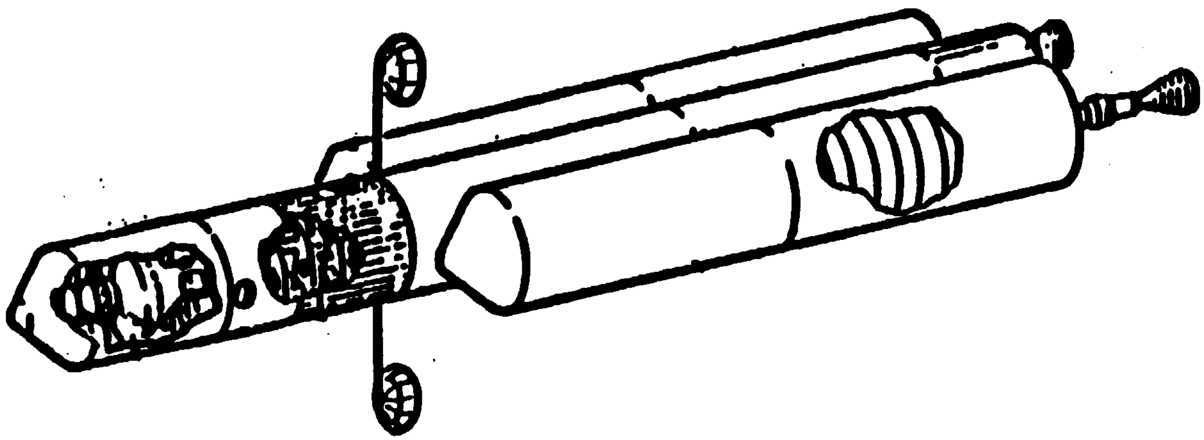
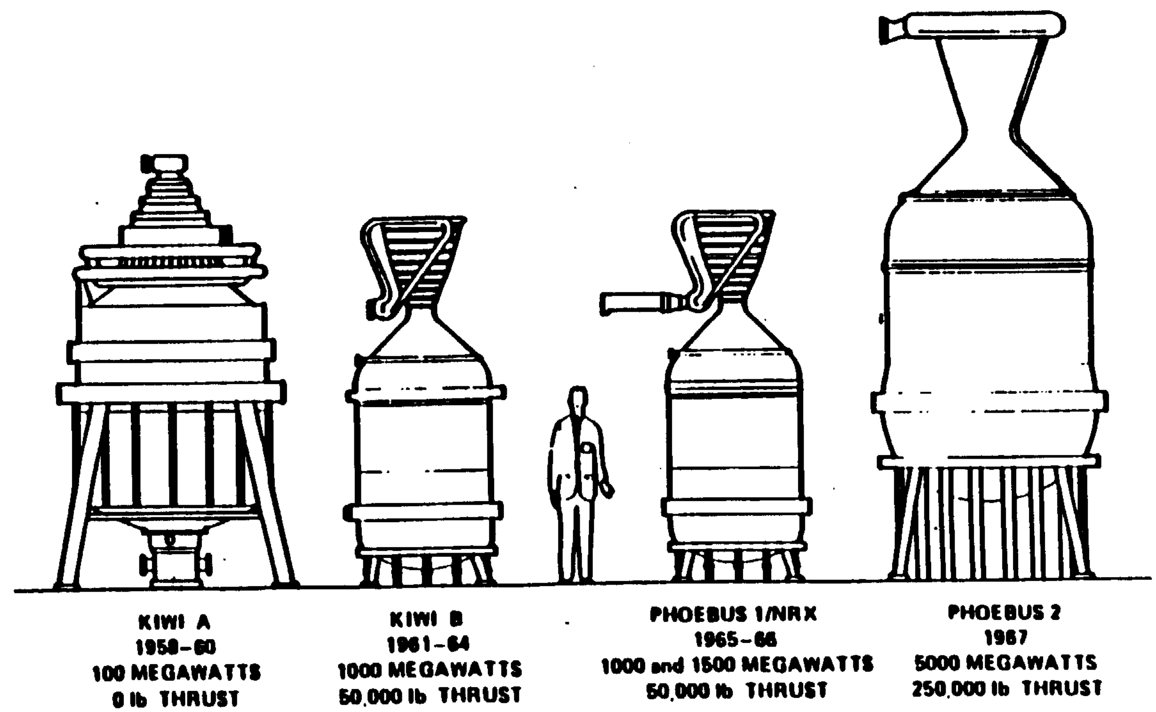
Benjamin Donahue

Early NTR Concepts

Initial nuclear rocket development was a joint Atomic Energy Commision-USAF project that started with exploratory research in 1953 bearing the name 'ROVER'. Soon to become a AEC-NASA program, early research was transformed into test hardware in July 1959 with the KIWI A test reactor (100 MWt, 0 lbf thrust) and peaking with the Phoebus 2A test reactor in 1967 (5000 MWt, 250k lbf thrust, 840 s Isp). In 1961, development began on the NERVA (Nuclear Engine for Rocket Vehicle Application) fight configuration series of full-up engines. Before termination of the NTR program in 1973, record performances of 62 minutes at continuous full power (NRX-A6), Peak fuel temp 2750 K (PEWEE), peak fuel power density 5200 MW/m3 (PEWEE) and 28 single engine restarts (XE) capability, among other achievements of an extensive test program, were seen at the Jackass Flats, Nevada test range. A total of 20 reactors were designed, built and tested between 1955 and 1973 at a cost of approximately 1.4 billion before support for the program ended. Post-Apollo plans for manned expeditions to the planets were abandoned due both to major cuts in NASA's budget and its transition of focus to development of a space shuttle. In the manned Mars mission plans of the 1960's NTR propulsion was the system of choice. The vehicle sketch shown below is of a NERVA-powered Mars spacecraft presented to a US Senate committee by Dr Werner von Braun in August 1969. Much technical progress was realized in the areas of reactor reliability and safety, as well as in all phases of reactor/engine/component integration. The NERVA design specifications are retrievable, down to the actual subsystem component design drawings. Had the program continued past 1973, the next step would have been the development and testing of a flight qualified engine, with a most probable application as an upper stage propulsion system for the Saturn launch vehicle. Recent technology advances since the early 70's, especially in the areas of fuel element/coatings materials improvements and fabrication techniques would provide significant performance gains without the need for reactor redesign. Applications of the NERVA type engines for Mars missions vehicles is presented in the following charts. An important emphasis of this section of the study has been laid on validating the performance gain that can be expected for 1990-2010 technology NERVA derivatives.

Early NTR Concepts





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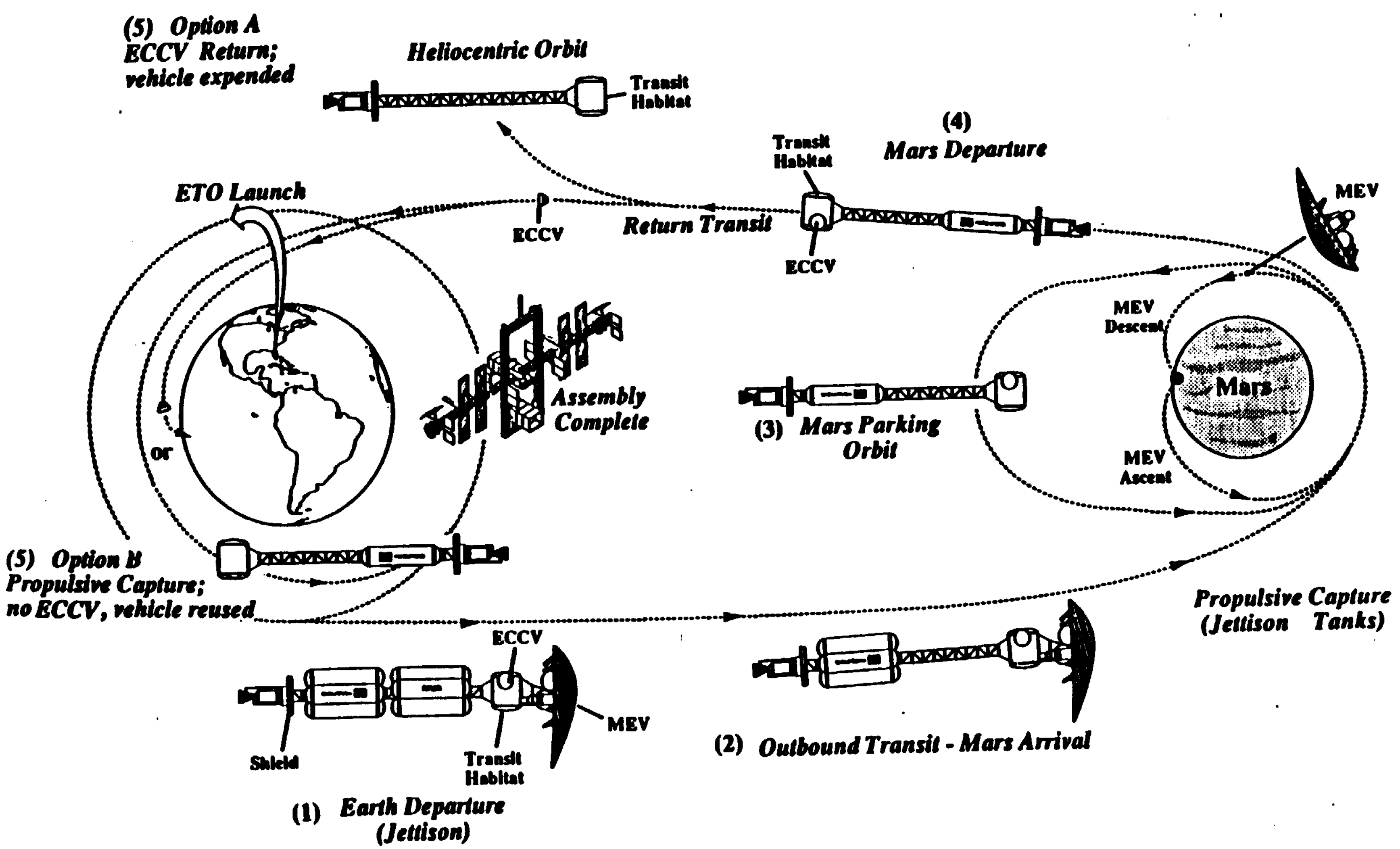
NTR MISSION PROFILE

Once assembled in LEO at Space Station Freedom the 2016 Mars NTR-powered Mars vehicle departs Earth utilizing a 2 burn TMI (Trans Mars injection) departure with its single 75k lbf thrust NERVA derivative engine and thus begins its 434 day journey. The payload shown consists of a 77 t MEV and a 4 man 32 t MTV crew hab module. A single Earth departure burn would incur sizable finite burn losses ('g-losses') due to the vehicles low overall thrust to weight ratio (vehicle T/W=0.04). Splitting the departure burn into 2 phases and firing each time near the orbit periapsis point is to be used to decrease these g-losses losses to an acceptable level.

After TMI the empty Earth departure Hydrogen propellant tanks are jettisoned as a means of lighting the load for all subsequent burns. Orbit capture at Mars is done all propulsively and as before, propellant tank(s) are jettisoned after the burn. After the surface stay and crew return to the transfer vehicle via the MEV ascent stage, the vehicle does a single Trans Earth injection (TEI) burn and begins the inbound journey with only the MTV crew module as payload.

After a inbound midcourse correction burn the vehicle will return to Earth one of two ways, shown on the sketch as option 5a or 5b. 5a is the vehicle expendable mode - the crew enters a small Apollo type Earth Crew Capture Vehicle (ECCV) weighing about 7 tons which enters the atmosphere via heat shield aerobraking to achieve eventual splashdown in the Pacific. Option 5B is the vehicle reuse mode - the vehicle does an all propulsive Earth capture burn into a 500 km by 24 hr elliptical orbit which allows reuse of the vehicle for a later mission. No ECCV is taken for this latter case.

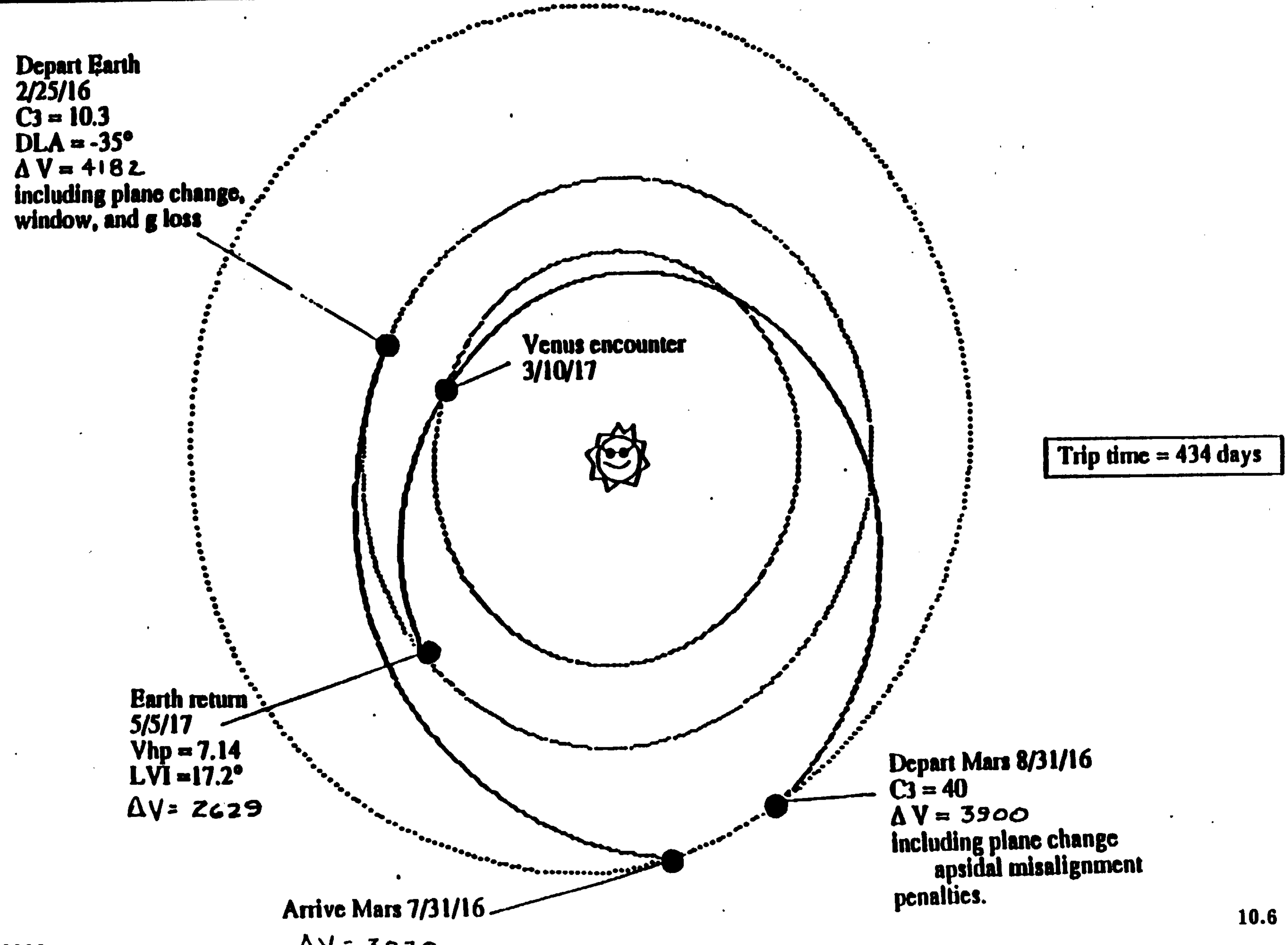
2016 NTR Vehicle Mission Profile



2016 NTR FLIGHT TRAJECTORY

The flight trajectory shown below was utilized as the trajectory from which all the NTR vehicle propellant loadings were based. The mission delta V's are listed for the TMI, Mars arrival, TEI and Earth arrival burns. Not listed are the outbound and inbound midcourse correction delta V's which are 120 and 90 m/s respectively. This 434 day trajectory is characterized by a 30 day Mars stay time and the inbound Venus swingby assist and was considered as a near optimum case for the 2015-2016 NTR vehicle design case study.

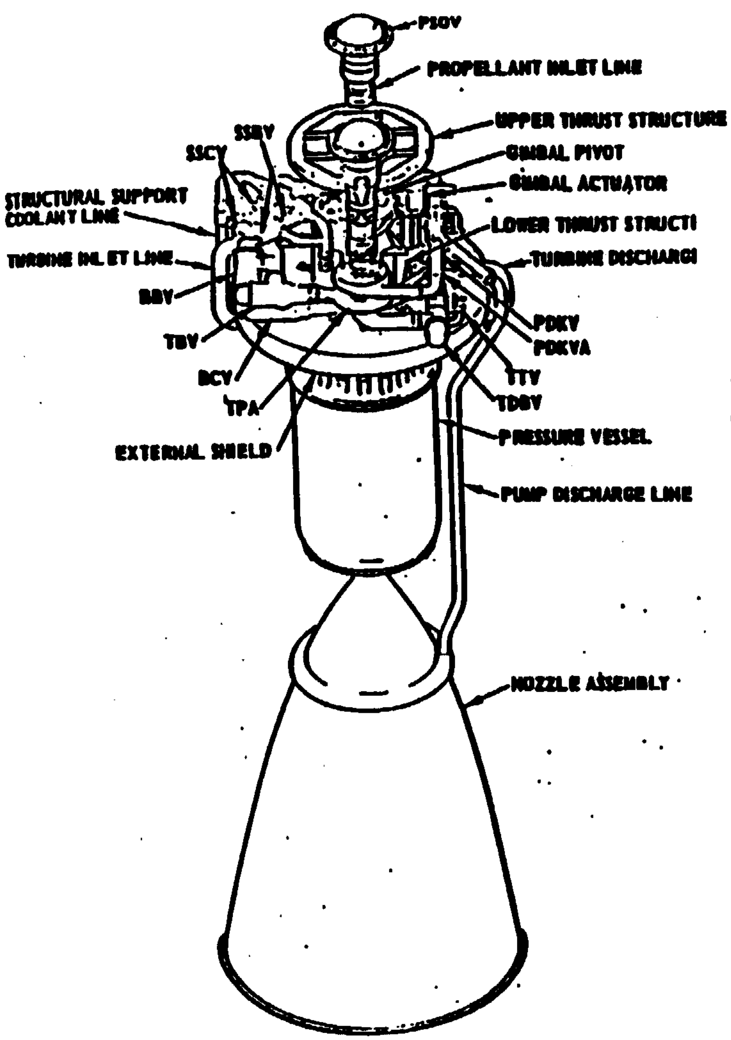
2016 NTR Reference Trajectory
Boeing #2 Modified/ 464 day / Venus swingby

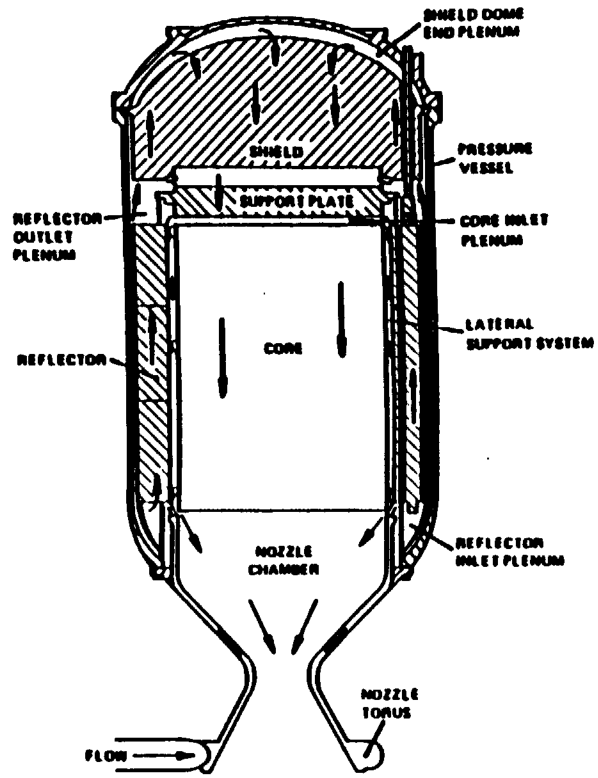


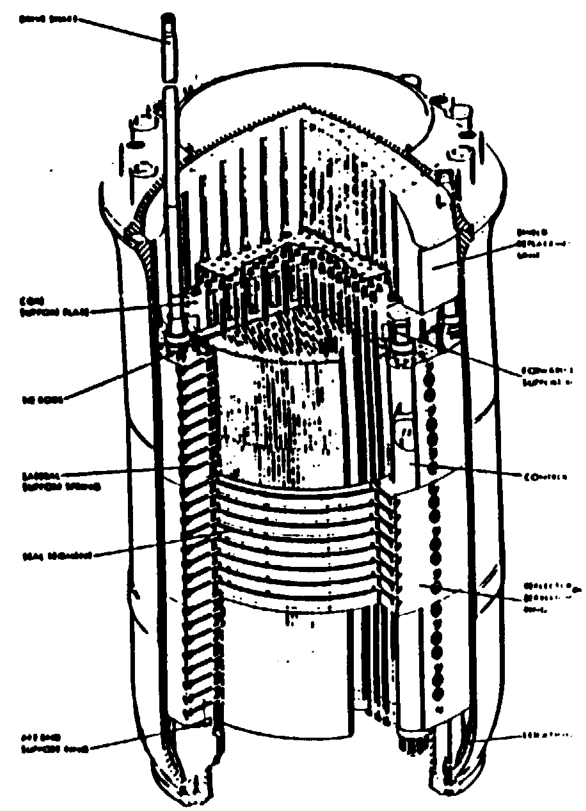
70'S NERVA ENGINE

The sketches below are of the NRX class NERVA engines used as a departure point in the NTR propulsion analysis and trades

70's NERVA Engine





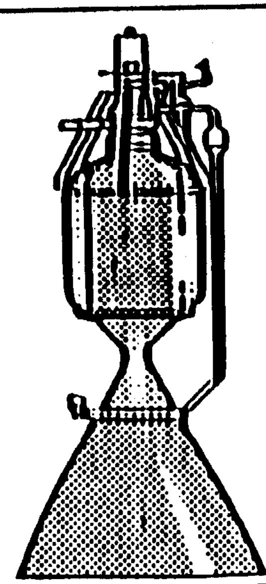
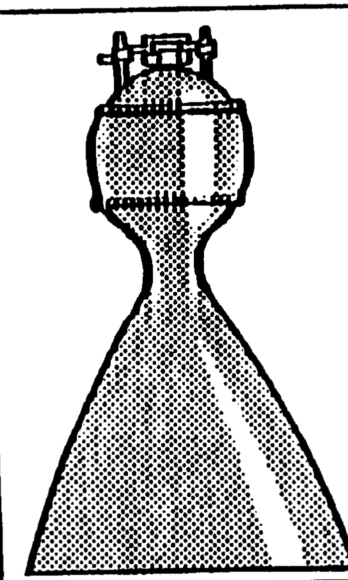


Advanced Propulsion System Characteristics

Two enhanced NERVA technology engine system characteristics are listed along with a radial flow low pressure reactor engine design for comparison with the demonstrated NERVA operating characteristics typical of the late 1960's.

Advanced Propulsion

System Characteristics

System	Solid Core Nuclear Thermal			
	Axial Flow (NERVA)			Radial Flow
Engine	Demonstrated	Advanced	Low Pressure	Low Pressure
Thrust	75000 lbf	75000 lbf	1500 lbf	5000 lbf
Isp	850 sec	925 sec	1100 sec	1250 sec
Pressure	450 psi	450 psi	15 psi	7 psi
Temperature	2250 °K	3000 °K	3200 °K	3500 °K
T/W	3	6	0.5	2
Description				

NTR Solid Core Fuel Element Temperature and Endurance Limits

(1) Fuel element temperature limits
ref: *Nuclear Space Propulsion*, H. F. Crouch, 1965

UZrC is the preferred ternary fuel for temperature and nuclear reasons. Its temperature advantage over that of UNbC (within the range of interest) is evident in the figure. The UTaC system is attractive at the less than 50 % UC content, but its neutron absorption cross section is disadvantageous. The melting point of UZrC ranges from a low of 5450 (F), to a high of 6100 (F)[3644 (K)]. For the selection of a 50% UC content [melting point~5850(F)/3505(K)], a typical reduction of 500(R)/278(K) (allowed to provide the required strength) would dictate an approximate maximum fuel element operation temperature of 5350(F)/3227(K)

(2) Fuel Element Endurance
ref: *Space Nuclear Power*, J.A.Angelo & D. Buden, 1985

The figure illustrates the anticipated lifetimes at various operating temperatures for graphite, composite and carbide fuel elements. If a ten hour life is desired, the reactor would have to operate around 2200-2300 K with a graphite matrix fuel, for composites approximately 2400 K, and for carbides possibly as high as 3000 K.

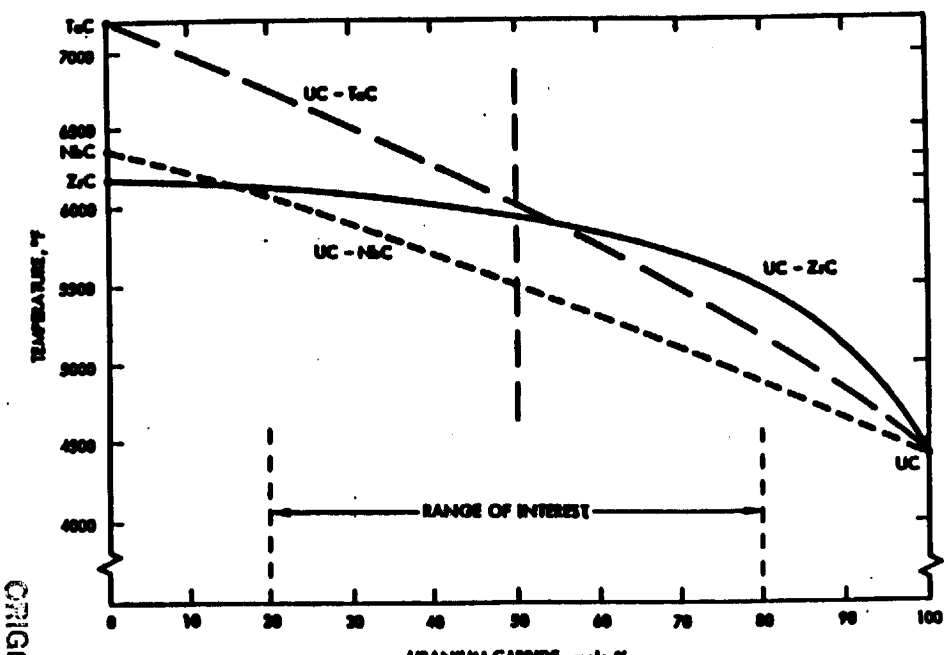
NTR Solid Core Fuel Element Temperature and Endurance Limits

ADVANCED CIVIL SPACE SYSTEMS

BOEING

Fuel Element Temperature Limits for Carbides

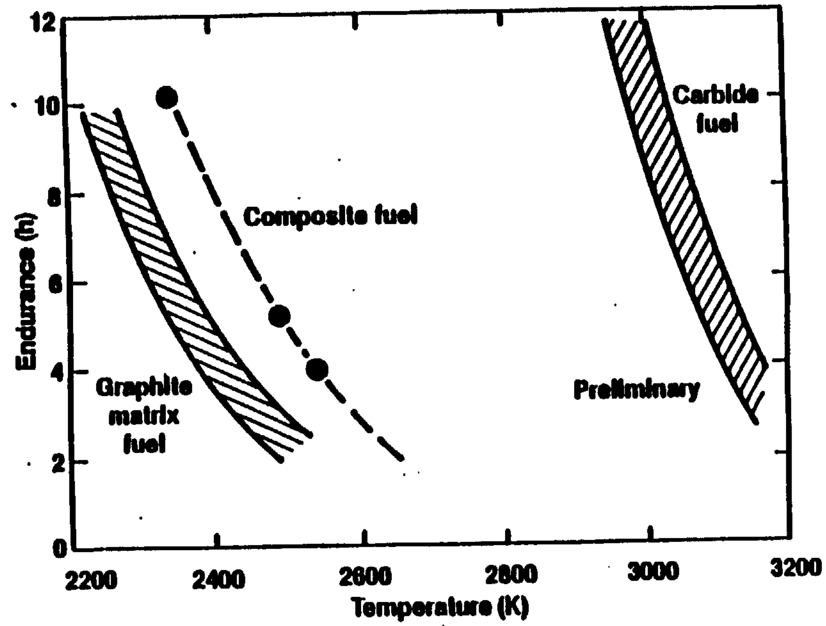
melting points of Ternary (=3 components in a solution) Carbide (=inorganic compound such as metal or ceramic with carbon) fuels vs UC mole % for 3 ternary systems which are satisfactorily stable with uranium:U-Ta-C,U-Nb-C, U-Zr-C;(Ta=tantalum,Nb=niobium,Zr=zirconium)



Source: *Nuclear Space Propulsion*, Holmes F. Crouch,1965

Fuel Element Endurance

Comparison of projected endurance of several fuels vs coolant exit temperature Graphite,Composite and Carbide fuels



David S. Gabriel, *Statement to Committee on Aeronautics and Space Sciences*, US Senate, 1973

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NTR Vehicle Isp as a Function of Hydrogen Temperature and Pressure

(1) Isp as a function of chamber temperature

Isp is proportional to the square root of chamber temperature. The figure on the left illustrates the rise in Isp with temperature for a range of 2600 to 5000 K. The UZrC ternary carbide fuel elements appear to have a upper operating temperature limit of around 3200-3300 K, given a choice of a operating temperature margin (typically about 278 K) from the melting point of 3590 K at a 40 % UC content. Once such a material limit has been reached, an additional gain in Isp can only be achieved by lowering the chamber pressure. The motivating force behind operating at lower pressures (and at these high temperatures) is the marked increase in the percentage of the H2 gas that disassociates into atomic hydrogen (see next chart entitled 'Hydrogen Disassociation'). Disassociation with accompanying recombination in the exhaust nozzle provides a significant kick to Isp.

A family of constant chamber pressure lines illustrate the theoretical gain in Isp that can be expected for lower pressure systems. This data is from a 1960 NASA report and is the basis for Idaho National Engineering Laboratories (INEL) analysis of conceptual low pressure reactor designs as a means for performance increases beyond NERVA derivatives. The theoretical Isp improvement, as indicated solely from this data, would be approximately 200 sec (1250 vs 1050 sec) for a 10 psia system vs a 450 psia system, if a 3200 K chamber temperature was maintained for both. The certainty of seeing this magnitude of improvement in actual practice is has yet unproven, and certainly has questions that might never be resolved until an actual reactor is tested - it would require a reactor design radically different from NERVA. Such a reactor concept specifically tailored to take advantage of disassociation at low pressures has been put forward by the INEL team; an illustration is given in the chart 'Distinctions Between Low Pressure NTR Reactor Concept and Demonstrated NERVA Reactor'.

(2) Isp as a function of chamber pressure

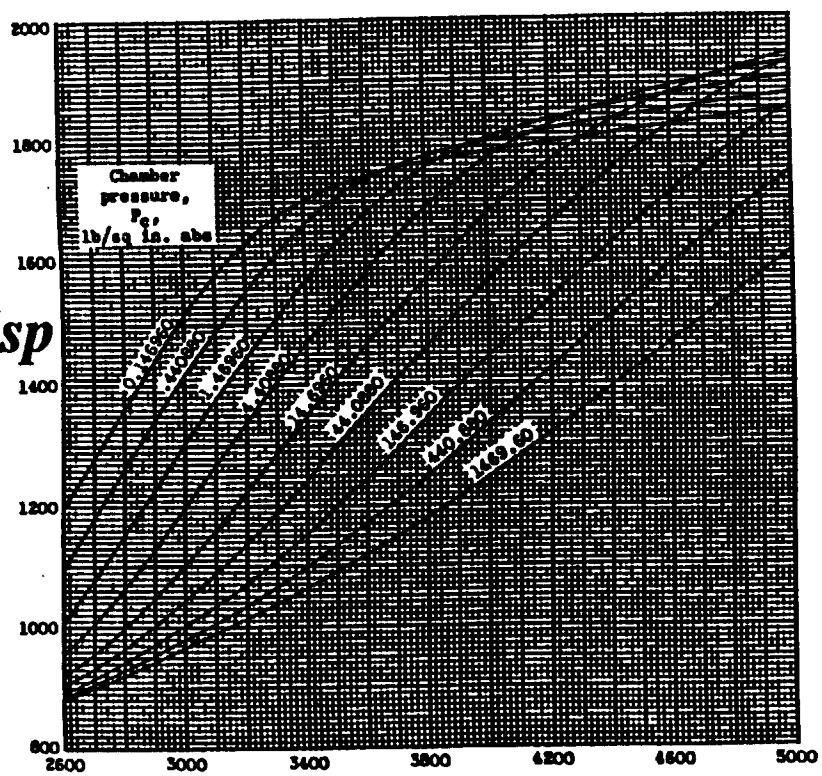
The same data as above, plotted vs chamber pressure on the x axis.

NTR Vehicle Isp as a fuction of Hydrogen Temperature and Chamber Pressure

Isp as a fuction of chamber temperature

Source:

Compilation of Thermodynamic Properties, Transport Properties, and Theoretical Rocket Performance of Gaseous Hydrogen by Charles R. King, NASA/LeRC, NASA TN D-275 April 1960

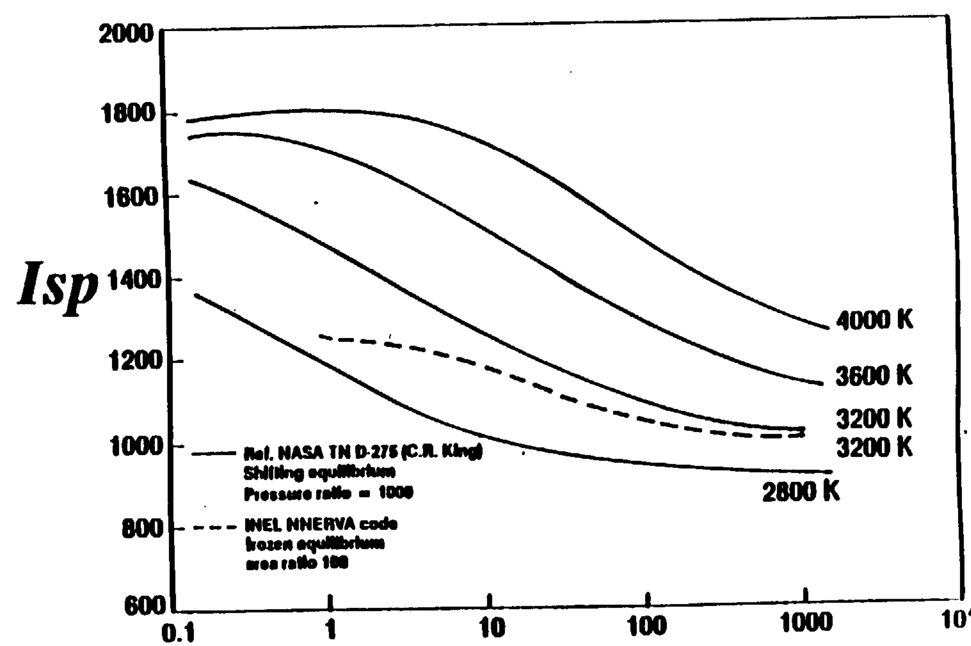


Isp in vacuum for gaseous normal hydrogen assuming equilibrium composition during an isentropic expansion to a pressure ratio of 1000

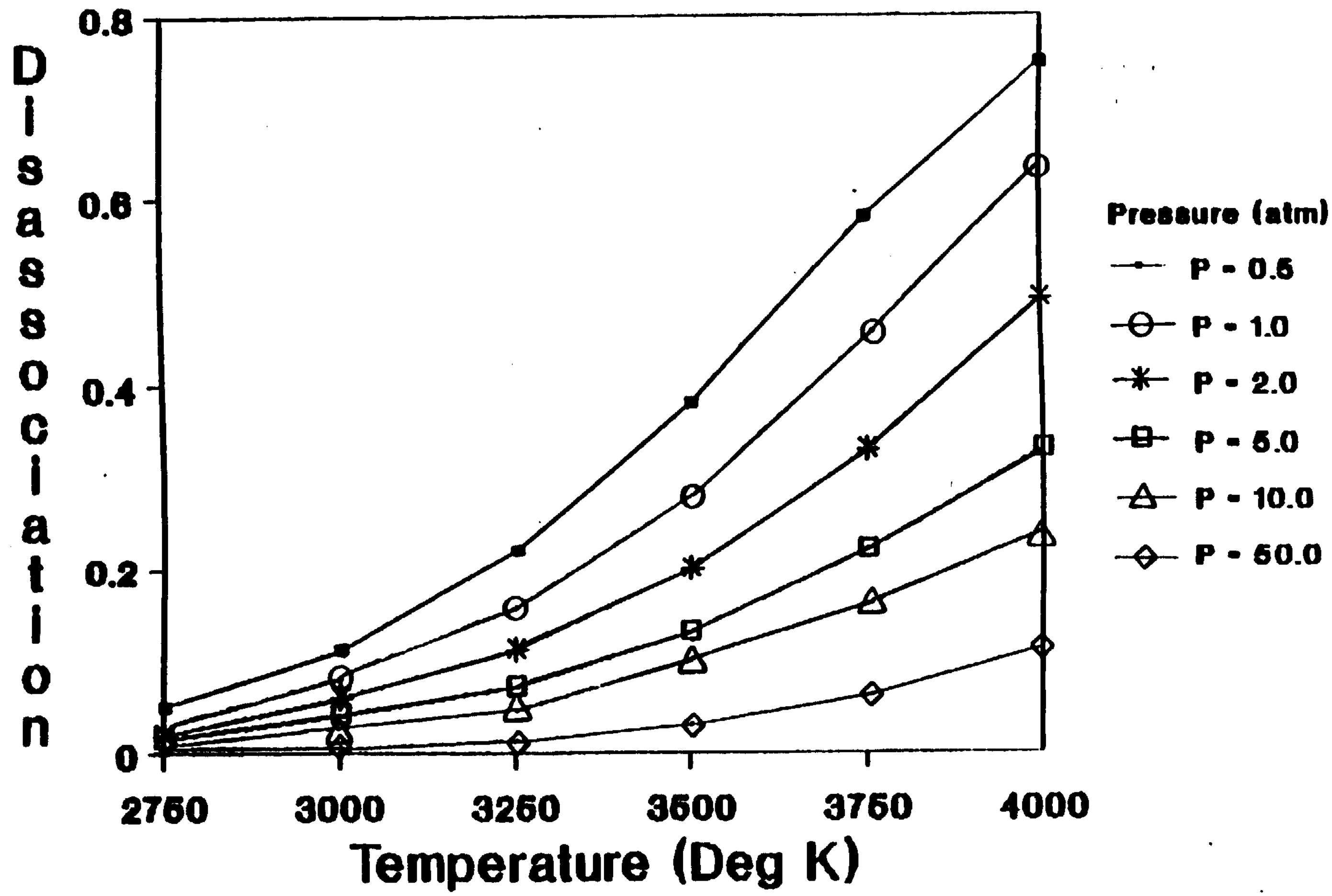
Isp as a fuction of chamber pressure

Source:

Pressure Fed Nuclear thermal Rockets for Space Missions briefing charts presented at NASA MSFC meeting 1989 J. Ramsthaler/C. Leyse, Idaho National Engineering Lab.



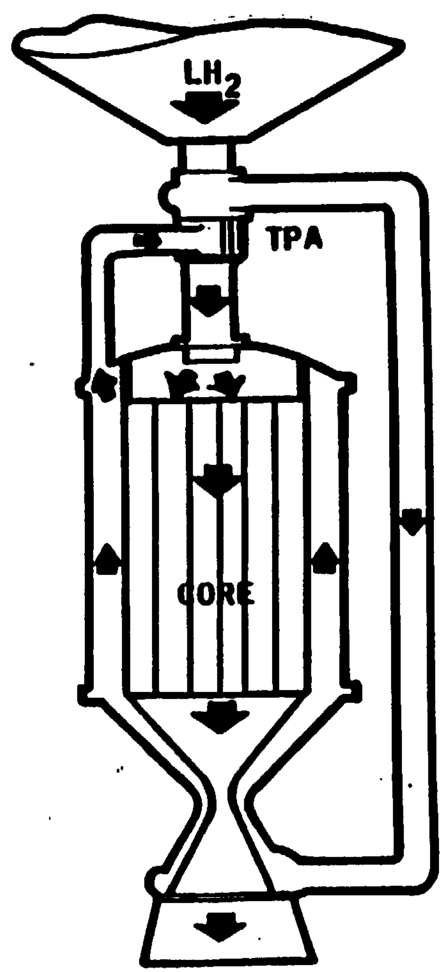
HYDROGEN DISASSOCIATION



Distinctions between Low Pressure NTR Reactor Concept and demonstrated NERVA Reactor

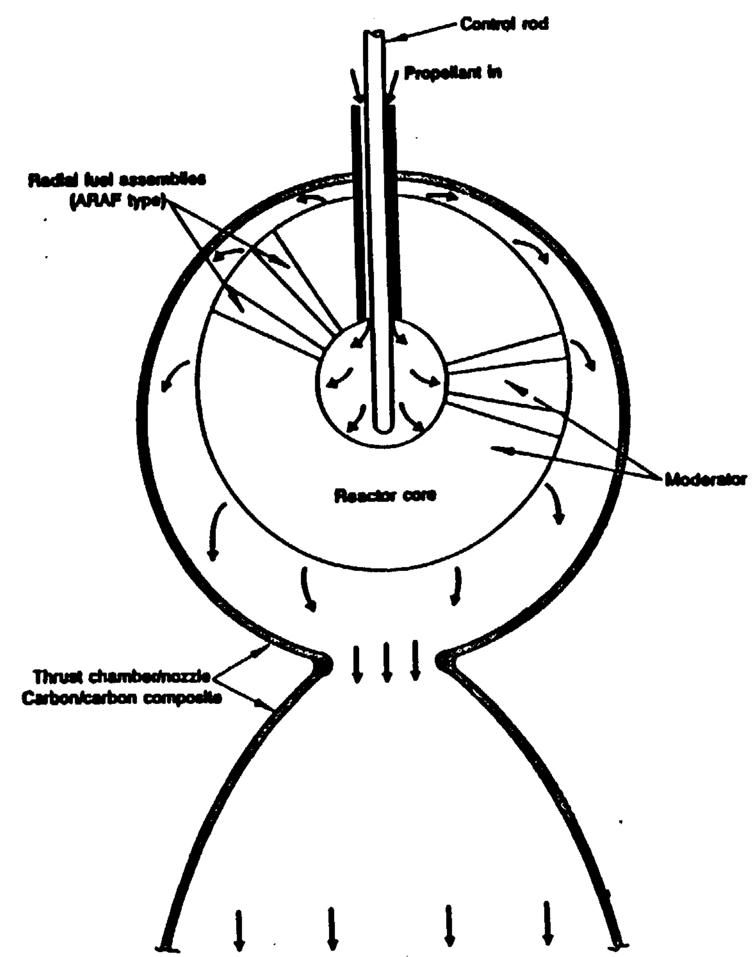
70's Axial Flow NERVA configuration

- *High chamber pressures: 450 psia*
- *Pump feed engine; topping cycle shown in diagram*



Radial outward flow Low Press reactor sys

- *Low chamber pressures: 10 psia and lower*
- *Pressure feed system*



2016 Mission Mars NTR Vehicle Comparison to Reference Chemical/aerobrake Vehicle

Five NTR propulsion Mars vehicle design concepts tailored to the 2016, 434 day mission trajectory are shown below, with pertinent performance characteristics, for comparison to the 90 day study chemical/aerobrake reference Mars vehicle modified for the 2016 434 day trajectory for a fair comparison. The vehicle concepts each represent a different engine performance level as described below.

(1) ***70's NERVA***: previously demonstrated NERVA technology of the late 1967 -70 era: example: Phoebus 2A demonstrated 840 isp in 1967

(2) ***Advanced NERVA***: modifications: higher temperature (2700 K) composite fuel elements, high expansion ratio nozzle

(3) ***Full potential NERVA***: modifications: maximum temperature (3100 - 3200 K) carbide fuel elements, high expansion ratio nozzle

(4) ***Full potential NERVA with MEV aerocapture into Mars orbit*** rather than all propulsive capture of joint MTV/MEV system as in cases 1,2 ,3 and 5. Aerocapturing MEV reduces MTV Mars propulsive capture payload and thus propellant. Disad: development of 2 major new technologies

(5) ***Low pressure reactor NTR propulsion***: completely new reactor concept: theoretical premise: very low pressure (10 psia or lower) at high temperatures (3200 K) produces significant H2 disassociation to get the improvement of Isp to 'beyond NERVA' 1250 sec plus levels.

The IMLEO statements at the bottom of the chart allow two return to Earth options:
(A) crew return via ECCV; vehicle expended - no reuse
(B) vehicle propulsive capture at Earth to a 500 km by 24 hr elliptical orbit for vehicle reuse

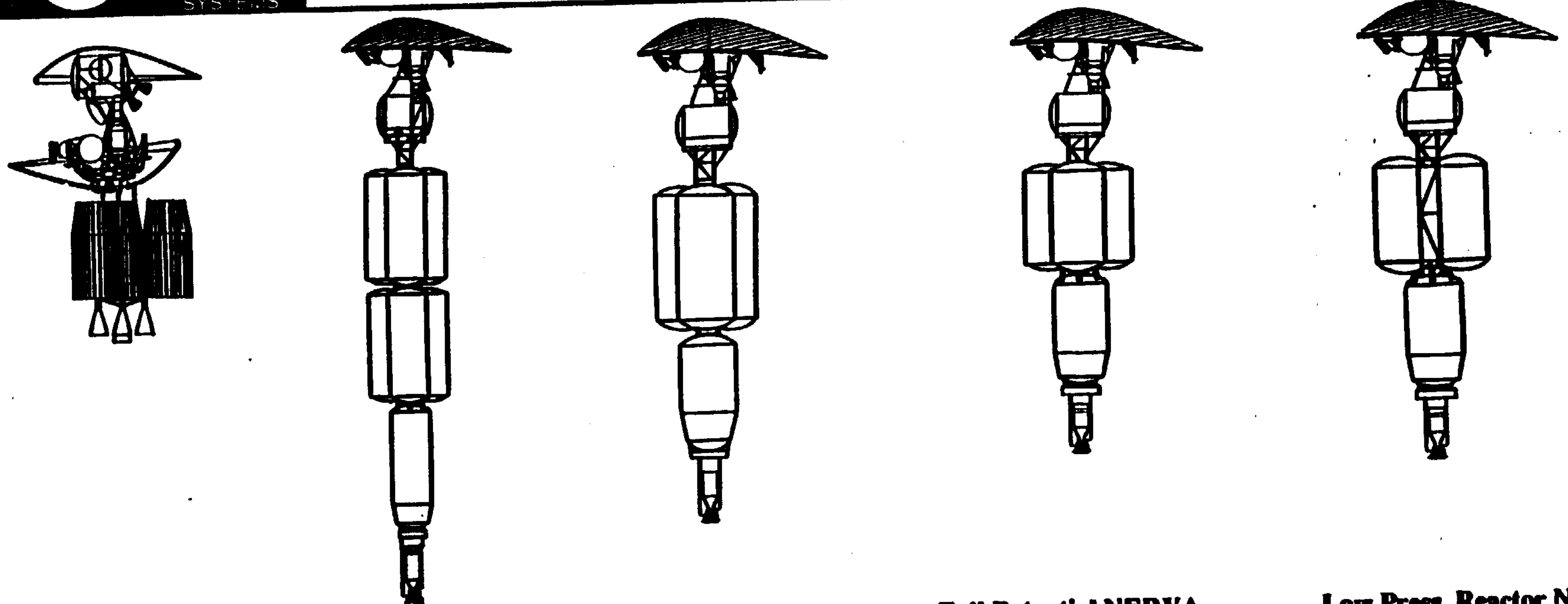
These NERVA and NERVA derivative vehicles all use single 75k lbf thrust engines. The LP vehicle has three 25k lbf thrust engines and all vehicles excluding chemical/aerobrake utilize multiperiapsis burn departures leaving Earth to reduce finite burn losses.

2016 Mission Mars NTR Vehicle Comparison to Reference Chemical/Aerobrake Vehicle

ADVANCED CIVIL SPACE SYSTEMS

Revision 4 March 30 1990

BOEING



Reference Chemical Veh	70's NERVA	Advanced NERVA	Full Potential NERVA		Low Press. Reactor NTR
Prop exit temp/Chamb press	2500 (K) / 450 psia	2700 (K) / 450 psia	3100 (K) / 1000 psia		3200(K) / 10 psia
Fuel element material	Graphite	Composites	Carbides with zirconium /carbide coating		Carbide w zirconium
Reactor perf. data reference	demonstrated	Stan Gunn/Rocketdyne	Stan Gunn/Rocketdyne		Ramsthaler/Leyse/INEL
Mars arrival: *tot aerocapt*	*full propulsive capt*	*full propulsive capt*	*full propulsive capt*	*Sep MEV aerocapt*	*full propul capt*
Trip time = 434 days	Trip time = 434	Trip time = 434	Trip time = 434	Trip time = 434	Trip time = 434
Isp= 475 MTV, 460 MEV	Isp = 850 sec	Isp = 925 sec	Isp = 1040 sec	Isp = 1040 sec	Isp = 1250 sec
Eng's: 4 x 200k (E dep)	One at 75k lbf	One at 75k lbf	One at 75k lbf	One at 75k lbf	Three at 25k lbf
Eng t/w,eng wt,tot shield wt	6/ 5.7 t / 4.5 t	6/ 5.7 t / 4.5 t	6/ 5.7 t/ 4.5 t	6/ 5.7 t / 4.5t	3/ 3.8 t / 4.5 t
E dep g-loss:100 m/sec	3 periapsis burn/200 m/s	3 periapsis/200 m/s	3 periapsis/200 m/s	3 periapsis/200 m/s	3 periapsis/200 m/s
Cooldown/tank fract/t dia	3% / 14% / 10.0 m	3% / 14% / 10.0 m	3% / 14% / 10.0 m	3 % / 14% / 10.0 m	3% / 14% / 10.0 m

IMLEO for ECCV return only ; vehicle expended

752 t	612 t	541 t	465 t	388 t	398 t

IMLEO for propulsive Earth capture; vehicle reuse mode

830 t	818 t	698 t	574 t	498 t	471 t

D615-10009

2016 Advanced NERVA NTR Reference Vehicle Configuration

The 925 Isp NERVA derivative engine was chosen by NASA MFSC as the reference propulsion system for the NTR vehicle studies. The performance of the 925 Isp system corresponds to a 'intermediate' reactor fuel element material. Composite fuel elements (see fuel element chart) operating such that the hydrogen propellant reaches approximately 2700 K at 450 psia chamber pressure would provide this Isp, given a large expansion ratio nozzle, and would require no redesign of the NERVA reactor beyond that necessary for integration of these higher temperature fuel elements (cooling and f element corrosion are such factors). An Isp of 925 is approximately 85 sec higher that that obtained by the Phoebus 2A reactor in 1967. Such a level of enhancement entails no high risk new technology development, rather it would be an extension of the advanced fuel element analysis that was already underway in the early 70's when the NERVA program was canceled - materials development and fabrication in general has seen a lot of advancement in the last 20 years. The reference vehicle was built around this performance level using the *Boeing Vehicle Synthesis Model*, a sophisticated computer code that outputs vehicle performance figures and weight breakdowns based on very specific vehicle configuration and requirements inputs too extensive to mention here. The vehicle, has illustrated has four 10 meter dia hydrogen prop tanks with a tank fraction of 14 %. 2 tanks for Earth dep prop that are jettisoned after TMI burn, one Mars arrival prop tank jettisoned after Mars capture and one tank that remains with the vehicle that holds both the Mars departure and Earth arrival propellant. A 2 m by 35 m SSF type truss is show nas connecting the inline Mars dep/Earth arr tank to the 33 t , 4 crew hab module and MEV. The MEV has only a *low energy, descent only* aerobrake - this is not a hign energy brake designed for Mars orbit capture. The vehicle does propulsive burns for orbit capture both at Mars and Earth. A summary weight statement of vehicle weight at burn ignition and propellant loads are also shown.

ADVANCED CIVIL SPACE SYSTEMS

Advanced NERVA NTR Reference Vehicle

10 meter dia propellant tank configuration

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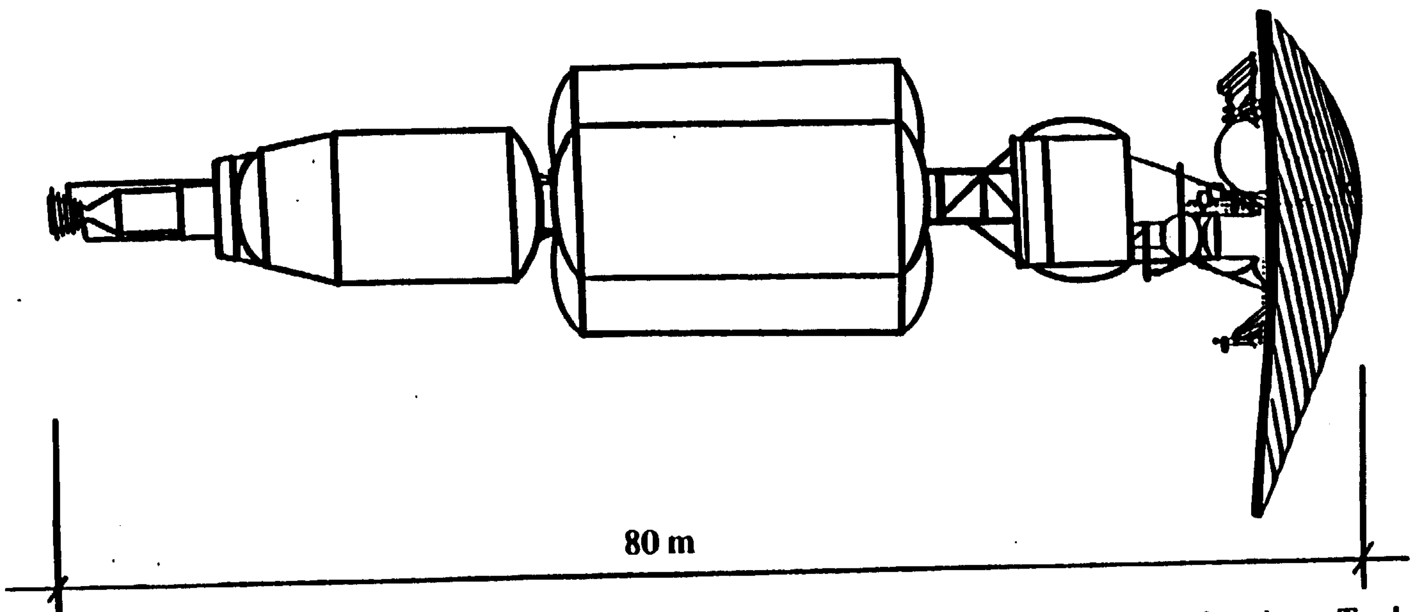
Engine	*Earth dep stg*	*Mars arr*	*Mars dep/E arr stg*
NERVA Technology Isp = 925 sec, Eng thrust = 75k lbf Eng wt= 5669 kg, Shield wt= 4.5 t	2 tanks, dia = 10.0 m length = 27.6 m	1 tank: dia = 10.0 m length = 29.6 m	1 tank: dia = 10.0 m length = 17.9 m

tank material: 4.0 mm continuous reinforced SiC/Al metal matrix; 2 m by 35 m truss



Stage	Mass at ignition	Burnout wt	Prop load	Tank set wt	Tank fraction	Tanks
Earth departure	697,856	432,134	265,722	40,888	13.3 %	2
Outb midcourse	391,246	387,384	3,862	-	-	
Mars arrivial	387,384	247,440	139,944	21,944	13.5 %	1
Mars departure	148,375	93,782	54,593	-	-	
Inb midcourse	93,782	92,558	1,224	-	-	
Earth arrival	92,558	66,219	26,339	13,950	14.5 %	1

2016 Advanced NERVA NTR Reference Vehicle Sensitivity to Isp and Engine T/W

The sensitivity of the reference NTR vehicle (IMLEO = 698 t) to changes to Isp is shown on the left with engine T/W held constant at 6 and 20.

The sensitivity of the reference NTR vehicle to changes in engine T/W is shown on the right with engine Isp held constant at 925 sec.

ADVANCED CIVIL SPACE SYSTEMS

2016 Advanced NERVA NTR Reference Vehicle Sensitivity to Isp & Engine Trust to Weight Ratio

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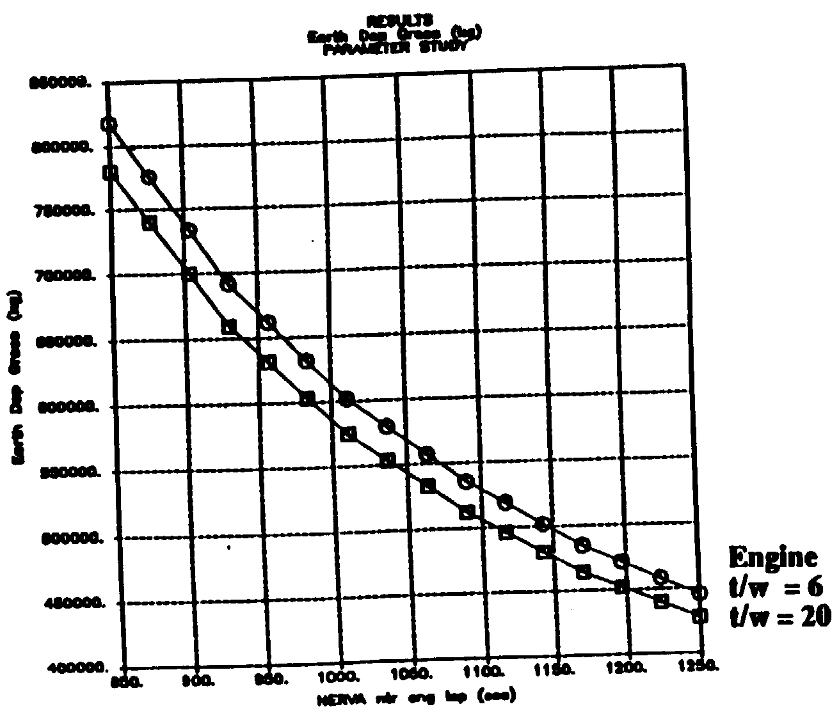
- 2016 Boeing #2 Modified Venus Swingby 464 day trajectory , propulsive capture at Earth, crew of 4
- Payload: MEV (77 t), MTV crew hab (32 t), 10.0 meter dia SiC/Al tanks - 14 % tank fraction
- One 75k lbf thrust eng, 4.5 t reactor shadow shield, eng wt & shield wt from NASA/LeRC propulsion task order

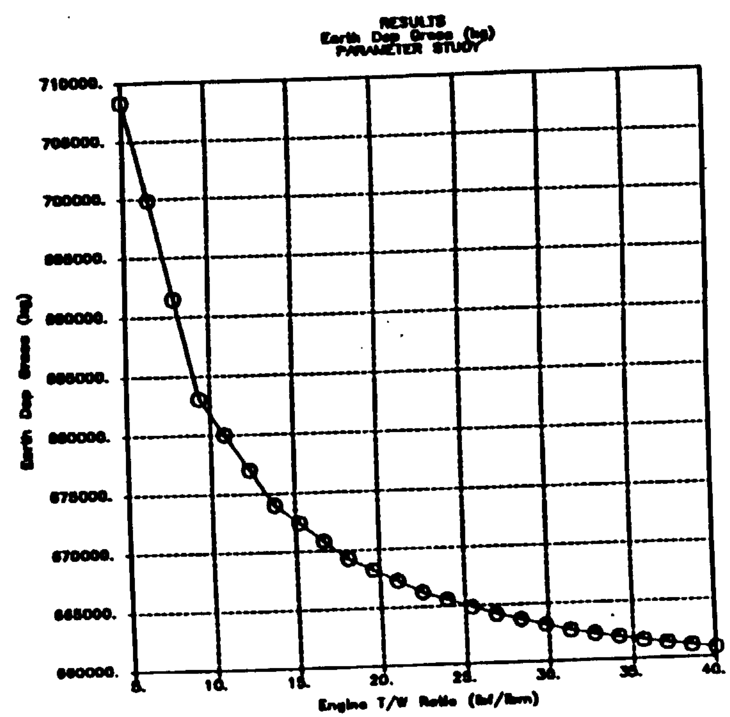
Vehicle IMLEO sensitivity to Isp

Top curve: Eng T/W held constant at 6:1, eng wt = 5669 (kg)
Bottom curve: Eng T/W held constant at 20:1, eng wt = 1700 (kg)



Vehicle IMLEO sensitivity to NTR Eng T/W

Engine Isp held constant at 925 sec



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Multiperiapsis Earth Departure Burns For Moderate to Low Vehicle T/W

ADVANCED CIVIL SPACE SYSTEMS

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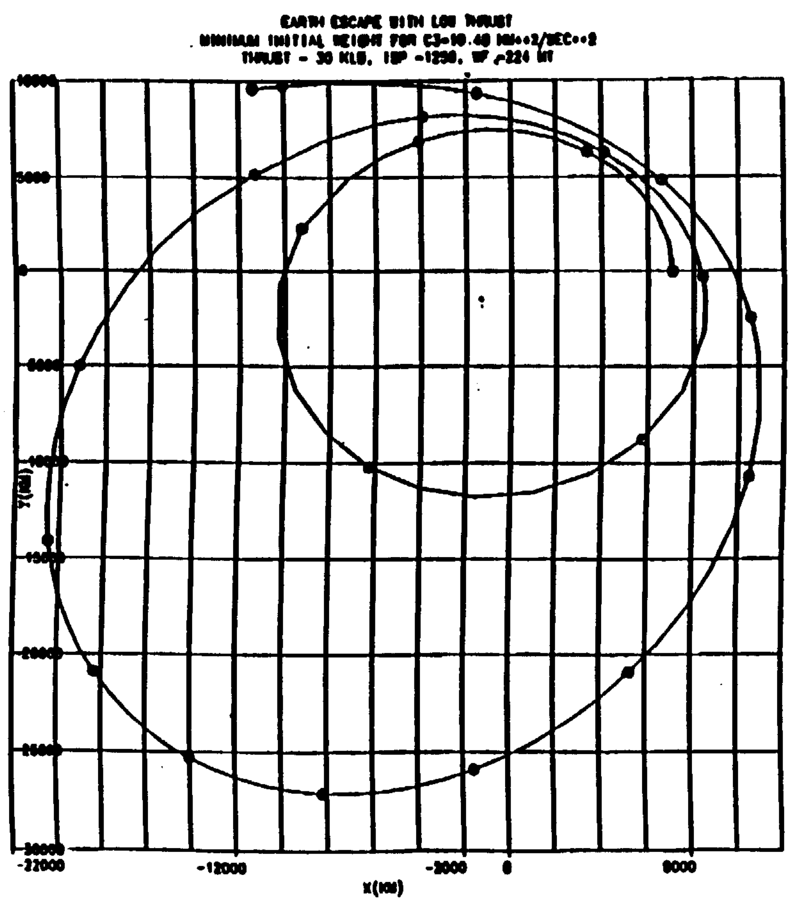
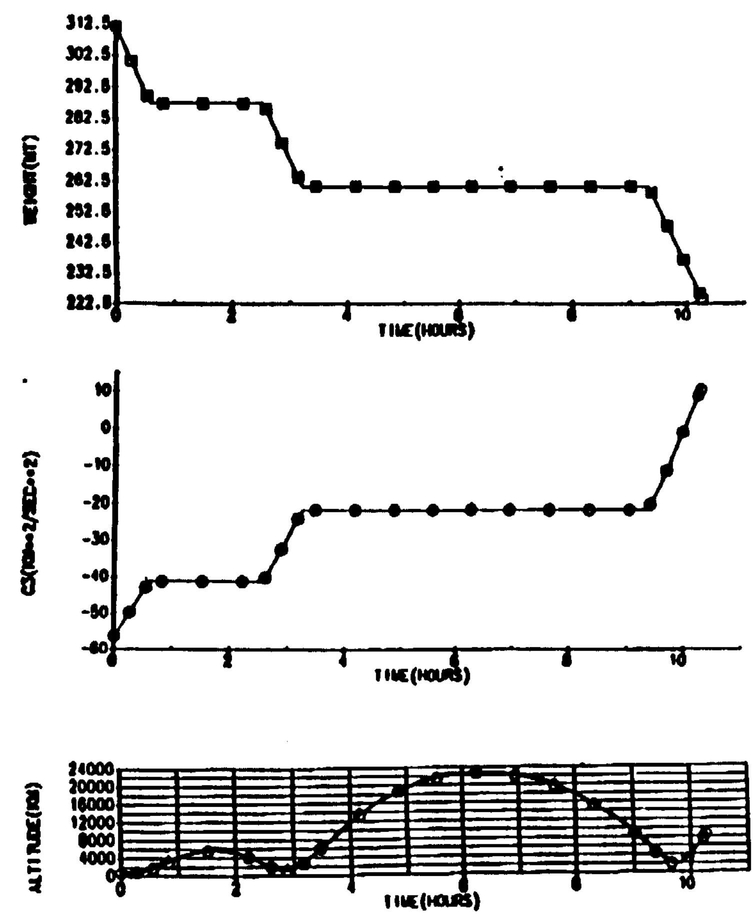
<u>Representative case</u> of low thrust 1250 Isp NTR sytem, 3 10k lbf engines Vehicle T/W = 0.04

3 burn Earth departure with 107 t MEV/MTV payload and 224 t Earth dep cruise mass; G-loss calc = 311 m/sec





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Nuclear Systems/Nuclear Solid Core Reactor Unique Issues

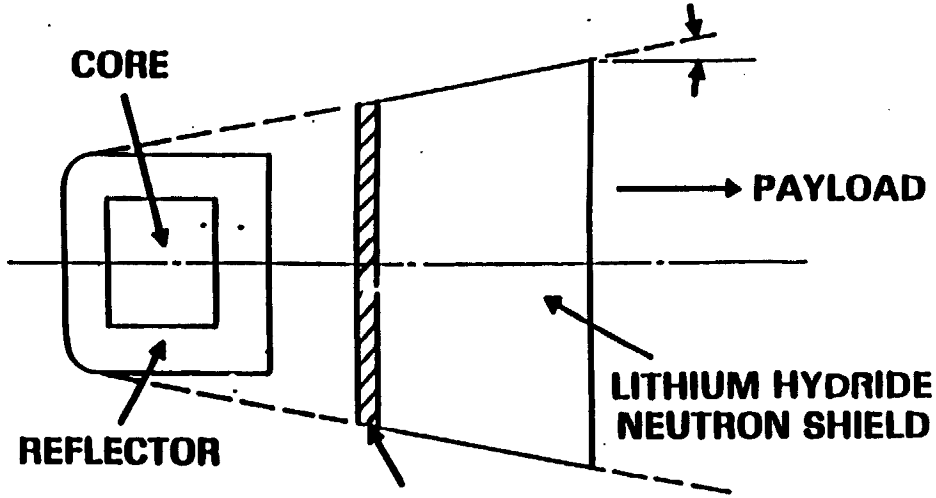
- Reactor radiation shielding
- Post burn reactor 'cooldown' requirement
- Current single NTR engine preference precludes traditional engine out margin
  - relatively heavy and costly reactor systems
  - multiple point source for radiation undesirable
- Full up engine test program concerns more complex now than NERVA era
- Very large hydrogen propellant tanks dominate vehicle physical configuration

NTR shadow shield configuration

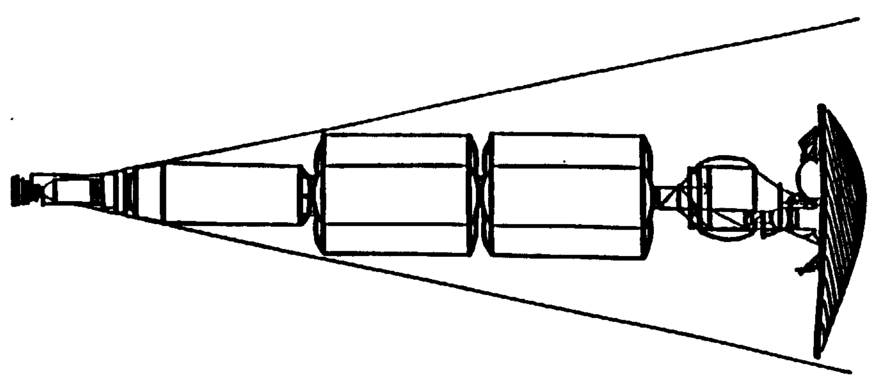
The NTR vehicle reactor shadow shield serves to shield the crew module and other structure (such as propellant tanks) from the high energy gamma radiation and the low energy thermal neutrons that are emitted from the reactor. A high density heavy metal material such as tungsten or lead serves to attenuate the gamma radiation while a material such as lithium hydride or water can be used to attenuate the thermal neutron flux. Minimizing the cone half-angle by configuration design is beneficial to minimizing the shadow shield size and weight.

NTR Shadow Shield Configuration

BOEING



Typical shadow shield configuration



NTR VEHICLE CONFIGURATION OPTIONS

The nuclear engine greatly influences the overall physical configuration of any NTR vehicle. The necessity for radiation attenuation between the engine source and the crew as well as the placement and staging of very large hydrogen propellant tanks are two major considerations that are unique to NTR systems. The following factors are applicable in this regard:

(1) Radiation dosage received by crew = 1/(separation distance) squared

Separation distance between the crew and reactor is a key player in reducing the amount of reactor generated radiation that reaches the crew habitat module. Since the reactor radiation dosage that eventually reaches the hab module is equal to the inverse of the separation distance squared, grouping the lengthy propellant tanks into a axial alignment rather than a radial cluster maximizes radiation attenuation by maximizing the separation distance provided by the tankage/structure without unduly penalizing the vehicle with structure dedicated solely to extending separation distance. Doubling the separation distance reduces the received dosage by a factor of 4.

(2) Axial alignment of tanks rather than radial clustering also allows the reactor radiation shadow shield protected cone half angle to be smaller since there would be less projected tank area around the reactor that could scatter direct radiation and thus become a secondary source. Any reactor shadow shield would include a very dense layer of material such as tungsten dedicated solely to gamma ray attenuation. Minimizing the shield size in important in keeping the weight down.

(3) Axial alignment provides more hydrogen propellant to be utilized as a secondary thermal neutron shield in the direct line between the crew cab and the reactor.

The configurations shown below are representations of various tank size and tank placement options. It is beneficial from a shielding viewpoint to keep the Earth arrival propellant in a 'inline' tank just behind the reactor shield. It is beneficial from an IMLEO weight standpoint to:

(a) jettison the tanks after each burn
(b) use as large a tank size as the launch vehicle(s) can deliver
(c) use advanced materials such as metal matrix composites to keep the tank fraction as low as possible

Other Issues include: Providing for tank release and jettison, minimizing and facilitating on orbit assembly, anticipating meteor shielding requirements (with or without a protection hanger at SSF), vehicle return for reuse refurbishment/resupply issues, artificial g accommodations and others.

ADVANCED CIVIL SPACE SYSTEMS

NTR Tank Sizing & Staging Options

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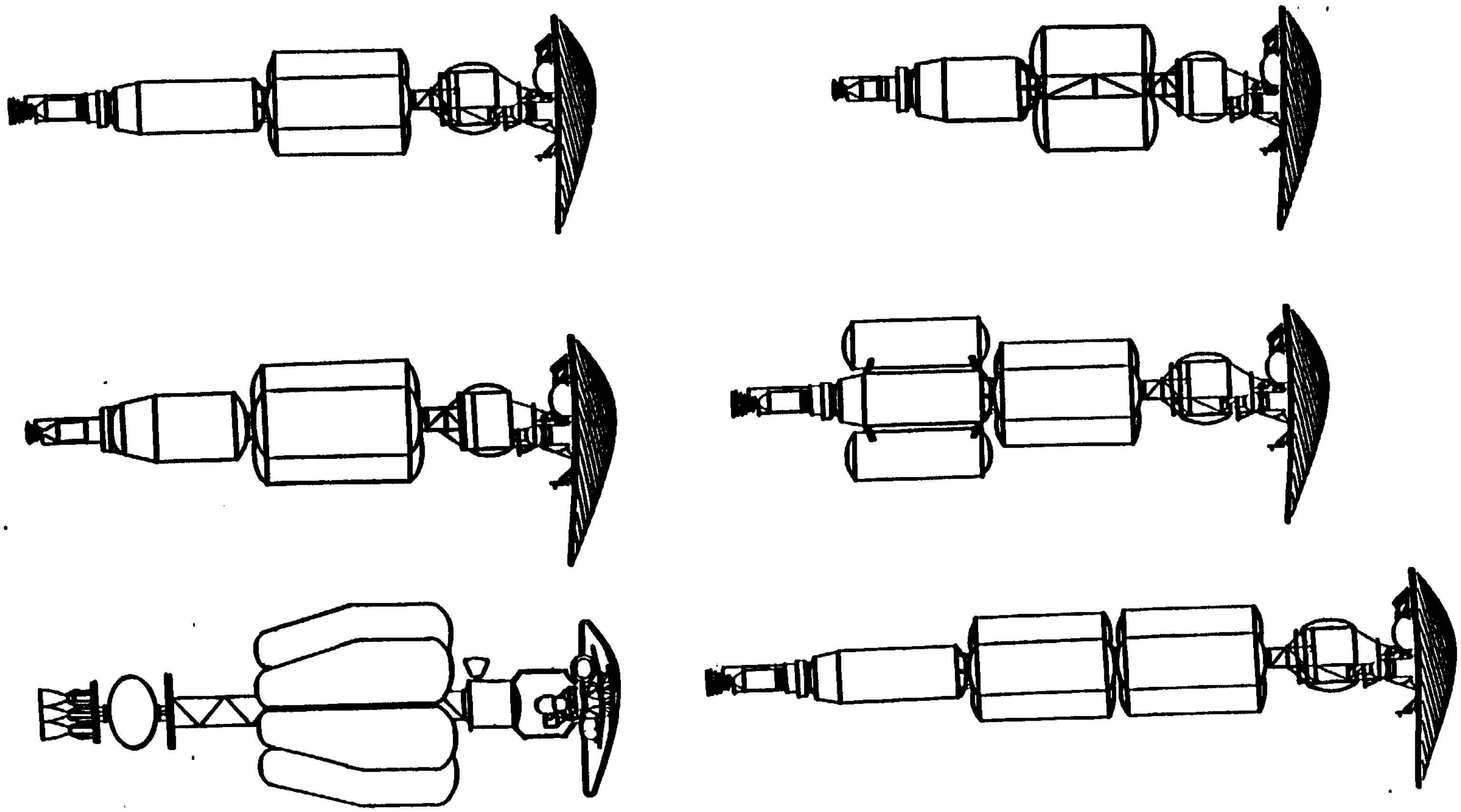

Mars NTR Reference Vehicle Weight Statement

The following six charts form a complete weight statement for the reference advanced NERVA NTR vehicle

Mars NTR Reference Vehicle: MEV Ascent Cab

STCAEM/bbd/13Mar90

crew of 4   3 day occupancy   synthesis model run number: marsntr.dat;100   2/27/90

Cab ECLSS	Atmospheric Revitization Sys/ Trace contaminant control assembly	123	CO_2 adsorption unit, expendable LiOH cartridge Pre & postsorbent beds,catalytic oxidizer for particulate & contaminant control
	Atmosphere Control System	62	Total & partial press control; valves,lines & resupply/ makeup O_2 & N_2 and tanks
	Atmos. Composition & Monitor Assem.	55	O_2 & N_2 monitor for ACS, particulate & contaminant monitor for ARS
	Thermal Control Sys	40	Temp control: sensible liq. heat exchanger, ext radiator wt included in 'secondary structure' mass
	Temp. & Humidity Control	240	Condensing heat exchanger, fans, ducting
	Water Recovery and Management	45	Stored Potable water only
	Fire Detection & Suppression Sys.	113	Automatic sys w manual extinquishers as backup
	Waste Management Sys and Storage	-	Considered part of 'Man Systems'
	Asc cab ECLSS mass	678	Apollo style open ECLSS system
Cab Structure	Primary/Secondary Structure	519	Overpressurized (20 psia) on launch for structural integrity. Stiffening rings added at cylinder/endcap interface for added strength. Skylab derived triangular grid floor with beam supports on 6" centers. Support ring interface on pressure vessel to carry loads imposed by the floor and equipment during launch to aerocapture.
	Berthing ring/mechanism (1)	139	
	Berthing interface plate (1)	90	
	Windows	90	
	Couches	80	
	Hatches (2)	80	
	Asc cab Structure mass	998	

all masses in kg

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Mars NTR Reference Vehicle: MEV Ascent Sys

crew of 4 3 day occupancy synthesis model run number: marsntr.dat;100 2/27/90

Ascent Cab		Structure	998	SSF dia center cylindrical section w ellip ends. Stiffening rings added. see 'Sructures pg'
		ECLSS	678	Open sys: CO2 adsorption unit, stored H2O, O2, N2, no Airl., no hyg w. see 'ECLSS pg'
		Command/Control/Power	330	Power: Fuel cells
		Man systems	82	Waste management sys/waste storage/medical equp.
		Spares/Tools	192	Subsystem component level spares
		Weight growth	376	15% growth for dry mass
		Asc 'dry' mass	2656	Total cab dry mass
		Consumables (food & water)	62	Minimum; food and water only; 3 day occupancy
		Crew/effects/EVA suits	760	crew of 4, 100 kg EVA suit per crew member
	22-1298	***Ascent cab gross mass***	3478	
Ascent stage inert	[45]	Fuel tank	437	2 LiAl cylindical tanks, 37k psi working stress, tank MEOP = 175 kPa
	[46]	Oxygen tank	444	2 LiAl spherical tanks, 37k psi working stress, tank MEOP = 175 kPa
	[48]	Vacuum shells	279	Scaled from tank surface area: 3 kg/m2 for every m2 of coverage
	[49]	Intertank	53	Scaled from Asc veh prop load: 0.3% of asc propellant wt (3 kg for every 1000 kg prop)
	[50]	MLI/Meteoriod Shield	303	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor shield: 2 (kg/m2)
	[500]	Main propulsion	564	2 x 30k lbf, low Pc, RL10 type eng's: Isp=460 sec, w extendible nozzles: AR = 200
	[525]	RCS inert	214	Estimate from RCS prop load
	[113,14]	Stage electrical/power	400	Fuel cells, batteries, vehicle & engine controls and monitoring sys
	[115]	Payload support	100	Structure
	[54]	Mass growth	419	15% growth for for inerts
	[55]	***Total Ascent stg inert***	3214	
Ascent prop & totals	[52]	Asc RCS propellant	211	Storable:N2O4/MMH propellant, Isp=280 sec, RCS dV=35 (m/sec)
	[57]	Asc usable Fuel	2508	LH2/LO2. Asc veh delta V= 5319 (m/sec) to 250 km periapsis alt., 1 sol orbit. Mixture
	[59]	Asc usable Ox	15165	ratio=6:1. Asc veh propellant refrigerated untill MEV separation from MTV
	[1298]	Surface samples	1000	
	[65]	***Asc veh at liftoff***	25458	
	[56]	Fuel boiloff on surf	71	30 day surf stay; boiloff calculations from Boeing's 'CRYSTORE' program based on MLI
	[58]	Ox boiloff on surf	98	thermal conductivity data from NASA studies performed by Lockheed
	[63]	***Ascent veh landed mass***	25626	1000 kg of surface sample mass shown as liftoff mass

all masses in kg

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Mars NTR Reference Vehicle: MEV Descent Sys

crew of 4   3 day occupancy   synthesis model run number: marsntr.dat;100   2/27/90

Desc stage inert	[98]	Desc Fuel tank	320	2 LiAl spherical tanks, 37 ksi working stress, tank MEOP = 175 kPa
	[99]	Desc Oxygen tank	339	2 LiAl spherical tanks, 37 ksi working stress, tank MEOP = 175 kPa
	[111]	MLI/Meteor shield	244	MLI:density=32 (kg/m3); 100 layers at 20 layers/cm. Meteor Sh.:2 (kg/m2)
	[501]	Desc main propulsion	1127	4 x 30k lbf RL10 type eng's: Isp=460 s, w extend/retract nozzles: AR = 400
	526+1273	RCS inert	366	Storable: N2O4/MMH propellant, Isp=280 sec, Asc RCS dV=35 m/sec
	[102]	Frame structure	1090	10% of desc stg inert mass + 2% of surf crew mod mass
	[103]	Landing legs	1643	3% of total landed mass
	[104]	Mass growth	769	15% growth for inert stage
	[105+104]	***Total desc stg inert***	5897	
Desc prop	[91]	Desc usable Fuel	1883	LH2/LO2. desc veh delta V= 931 (m/sec) from 250 km by 1 sol orbit.
	[92]	Desc usable Ox	11296	Mixture ratio=6:1
	[101]	Desc RCS prop	1417	Storable N2O4/MMH prop, Isp=280 sec, desc maneuver dV=100 m/sec
	sum	***Total MEV descent stg***	20494	
pay load	[77]	Surface crew mod	25000	Level II requirement: surface module, surf science and surf consumables
	[61]	Asc veh landed mass	24627	from "MEV Ascent System" page
	[78]	MEV lander aeroshell	7000	13% of MEV aerocaptured mass. Current (1/90) estimate of 19275 kg is based on structural analysis done after this reference wt statement was completed.
	[106]	***MEV mass at MTV sep***	77121	
		MEV landed mass	55525	Minus aeroshell, descent propellant and desc RCS propellant

all masses in kg

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Advanced NERVA NTR Reference Vehicle: MTV Transfer Crew Module

Crew Hab sys	[360]	Structure	9241	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
	[363]	ECLSS	4256	SSF derived with same degree of closure, sized for crew of 4 for 434 days
	[364]	Command/Control/Power	1159	Solar arrays, batteries , fuel cells and conditioning equip
	[368-316]	Man systems	4121	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
	[316]	Crew & effects	440	110 kg per person including personal belongings
	[373]	Spares/Tools	1496	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
	[247]	Radiation shelter	1802	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
	[377]	Weight growth	3107	15% weight growth for dry mass excluding crew & effects and radiation shelter
	[378]	***MEV 'dry' crew hab mod wt***	~~25022~~ 25622	Total MTV cab dry mass
	[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
	[371]	On board equip resupply	985	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
	[398]	Consumables	4417	Crew of 4 for 434 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
	[380]	***MEV crew hab mod gross***	31024	
Crew mod supp. sys	[i65]	Transfer science equipment	1000	Inb and outb MTV science hardware and supplies
	[i60]	TTNC & GN&C platforms wt	1150	4 platforms total
	[i59]	Communication satellites	0	flown on precursor missions
	i61+126	Artificial g tether mass	0	zero g environment
	[i57]	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
		MEV crew mod & support systems weight	33174	

Boeing vehicle synthesis model run number: marsntrmtv.dat;34 all masses in kg

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2016 Advanced NERVA NTR Reference Vehicle: Earth dep & Mars arr stages: $Isp = 925$

ADVANCED CIVIL SPACE — **BOEING**

Frame & propul	[159]	Spacecraft frame (truss) struct	4690	Truss struct: Graphite epoxy, Ec= 16 Msi, Den=0.06 lb/in3, 2 m by 35 m SSF type
	[i343]	Eng thrust structure	1000	Estimate for thrust ring/structure and attachment frame for eng,shield, tankage at rear
	[i83]	RCS inert wt	600	estimate scaled from RCS propellant load
	[709]	Main prop line wt	789	Main line from tank lines to reactor, L=35m,d wall s steel;dens=7833kg/m3,t=0.8mm
	[160]	Mass growth	1212	15% mass growth
	[518]	Engines wt (1)	5669	*75k lbf Thrust*, wt estimate: NASA/LeRC propul task order (Westinghouse, others)
	[543]	Engine shield wt (1)	4500	4500 lbf shadow shield wt from LeRC propul task order
	[118]	RCS prop wt	635	Transfer RCS dV = 20, Isp = 300, storable biprop
	[696]	***Frame & propul 'dry' wt***	19095	
Earth dep stg wt	+[i343]			
	[586]	Earth dep usable propel tot	253070	***Earth dep dV: 4182 m/s (includes 200 m/s gloss for 2 burn E dep) ;Isp = 925 sec***
	[701]	Earth dep prop residuals	5061	2% residuals/reserve left after boiloff, burn prop, and cooldown
	[697]	Earth dep burn 'cooldown' prop	7591	3% post burn prop for reactor cooldown: no thrust/Isp counted for this estimated %
	[705]	**Tot Earth dep stg prop load**	265722	***2 continuous reinforced Silicon Carbide /Al metal matrix tanks:*** dia: 10.0m,
	[668]	Single tank wt (cyl/ellip ends)	8592	L:27.6m, dens= 2436 kg/m3; 37ksi wk. stress, thickness = 4.0 mm, root 2 ellip ends
	[669]	MLI wt	2846	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x density
	[670]	Vapor Cooled Shield wt	2249	2 VCS: at 2 x 0.13mm Al outer sheets with 0.57 kg/m2 honeycomb core each
	[671]	Meteoriod shield wt	1909	One 0.80 mm sheet of Al; comparison: SSF uses 0.8 mm, Mariner 9 used 0.40 mm
	[i312]	Tank/frame attachment	600	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[672]	Tank feed prop line wt	159	Short prop line from tank to main prop line:double wall, stainless steel: 4 meter
	[674]	Mass growth wt	4089	25% wt growth for tank inert,MLI,VCS,meteor shield,prop lines, tank/veh attachment
	[588]	Sum of single tank inerts	20444	Total single tank inert wt;
	[592]	Total for 2 tanks	40888	Total for ' Earth dep tank set ': inert wt; Overall tank fraction [593] = 14.5 %
	[597]	***Earth Dep stage tot wt***	306610	Total Earth dep stg weight at time of Trans Mars Injection burn
Mars arr stg wt	[610]	Mars arr usable prop tot	130203	***Mars arr dV: 3870 m/s; eng Isp=925,*** H2 density = 70.8
	[702]	Mars arr prop residuals	2604	2% residuals /reserveleft after boiloff, burn prop, and cooldown
	[698]	Mars arr burn 'cooldown' prop	3906	3% post burn prop used for reactor cooldown; prelim;based on Westingh. estimate
	[611]	Mars arr stg outbound boiloff	3231	Boiloff for given MLI,VCS and Outb trip time; based on Boeing's 'CRYSTORE' prog
	[121]	Outbound midcourse prop	3862	Outb midc maneuver dV = 120 m/s; done by main propulsion from M arr tanks
	[612]	**Tot Mars arr stg prop load**	143806	***1 SiC/Al metal matrix tanks :*** dia:10.0 m, L:29.6 m, dens=2436 kg/m3;thick=4.0mm
	[633]	Single M arr tank wt	9217	
	[634]	MLI wt	3053	MLI: density = 32 (kg/m3); 200 layers (4 inches) at 20 layers/cm
	[635]	Vapor cooled shield wt	2412	2 VCS: at 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[636]	Meteoriod shield wt	2048	One 0.80 mm sheet of LiAl: assumption-LEO assembley in protective hanger
	[i312]	Tank/frame attachment	600	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[637]	Tank feed prop line wt	225	Double wall, stainless steel 4 meter H2 propellant line;dens= 7833 kg/m3, t=0.8mm
	[639]	Mass growth wt	4389	25% wt growth for tank inert,MLI,VCS,meteor shield,prop line&tank/veh attachment
	[614]	Sum of single tank inerts	21944	Total for single tank, with all inerts.
	[615]	Total for 1 tank	21944	Overall tank fraction [620] = 13.2 %
	[617]	***Mars Arr stage wt***	165750	wt at time of Earth departure

REVISION 4 *March 30 1990* all masses in kg

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ADVANCED CIVIL SPACE SYSTEMS

2016 Advanced NERVA NTR Reference Vehicle: Combined Mars dep & Earth arr stg: *Isp = 925 sec*

BOEING

Revision 4 March 30 1990

Propulsive capture at Earth for reuse

Mars dep & Earth arr stg wt

[128]	Mars dep usable prop load	50022	***Mars dep dV= 3900 m/s; eng Isp=925 sec,*** H2 density =70.8
[703]	Mars dep prop residuals	1001	2% residuals/reserve left after boiloff,burn and cooldown
[699]	Mars dep burn 'cooldown' prop	1501	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
[498]	Mars dep stg outbound boiloff	1722	Out b boiloff for given MLI & VCS insul.;no refrig,based on Boeing 'CRYSTORE'
[545]	Mars dep stg inorbit boiloff	347	31.5 day inorbit stay time
[122]	Inbound midcourse prop	1224	Inb midc maneuver dV=90 m/s; done by main propulsion system
[711]	**Tot Mars dep stg prop load**	**55817**	total at time of TMI burn
[561]	Earth arr stg usable prop tot	22387	***Earth arr dV=2629 m/s; propulsive burn capture into 500 km by 24 hr ellip orbit***
[704]	Earth arr stg prop residuals	448	2% residuals/reserve left after boiloff,burn and cooldown
[700]	Earth arr stg 'cooldown' prop	672	3% post burn prop for reactor cooldown; no thrust/isp counted in this approximation
[562]	Earth arr stg boiloff	2832	434 day b.off period; additional b.off from this tank also accounted in M dep p b.off
[563]	**Total Earth arr stg prop load**	**26339**	Total at time of TMI burn
[570]	**Total combined prop load**	**82156**	M dep/E arr prop: put in 1 tank along veh centerline aids NTR radiation attenuation
			1 continuous reinforced Silicon Carbide/Al metal martrix tank: dia:10.m, L:17.9m,
[683]	Single M dep/E arr tank wt	5694	filament wound; dens= 2436 kg/m3; 37ksi Wk. stress; tank skin thickness = 4.0 mm
[684]	MLI wt	1886	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x dens
[685]	Vapor cooled shield wt	1490	2 VCS - 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
[686]	Meteoriod shield wt	1265	One 0.80 mm sheet of Al; comparsion: SSF plans 0.8 mm, Mariner 9 used 0.4 mm
[i312]	Tank/frame attachment	600	Tank attachment mounting brackets, hardware and tank separation/release mechanism
[687]	Propel line/valves wt	225	length =10 m,double wall stainless steel H2 prop line; density= 7833 kg/m3, t=0.8mm
[689]	Mass growth wt	2790	25% wt growth for tank shell, MLI,VCS,meteor shield,prop line & attachment
[565]	Sum of inerts:single tank	13950	Total for single tank with all tank related inerts.
[566]	Total for 1 tank	13950	Overall tank fraction [571] = 13.3 %
[572]	***Combined Mars dep/Earth arr tank set & propellant load***	**96106**	Total for ' Mars dep/Earth arr tank set ' at time of TMI burn
[171]	***IMLEO***	**697856**	Boeing vehicle synthesis model run #: marsntrmtv.dat;55

all masses in kg

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NEP Update

Brad Cothran

ADVANCED
CIVIL
SPACE
SYSTEMS

Agenda

Mission Analysis
Nuclear Safety Operation Issues
Power System Mass Statement
Principle Findings

Optimum Mission Parameters for Various NEP Vehicles

Byrd Tucker of SRS performed this mission analysis under subcontract using POP (Parameter Optimization Program) and CHEBYTOP. Shown is initial mass in nuclear safe orbit (assumed to be 700 km) vs. Heliocentric Travel Time in days. Heliocentric travel time is equivalent to total manned trip time minus a 30 day stay time. Reference curves of the different vehicles are shown with corresponding power levels and specific masses (alpha's). The different vehicle alpha's were assumed early in the study to determine trends in the mission analysis. As the study progresses more accurate and detailed alpha's will be developed and incorporated into the mission analysis results.

ADVANCED CIVIL SPACE SYSTEMS

Optimum Mission Parameters for Various NEP Vehicles

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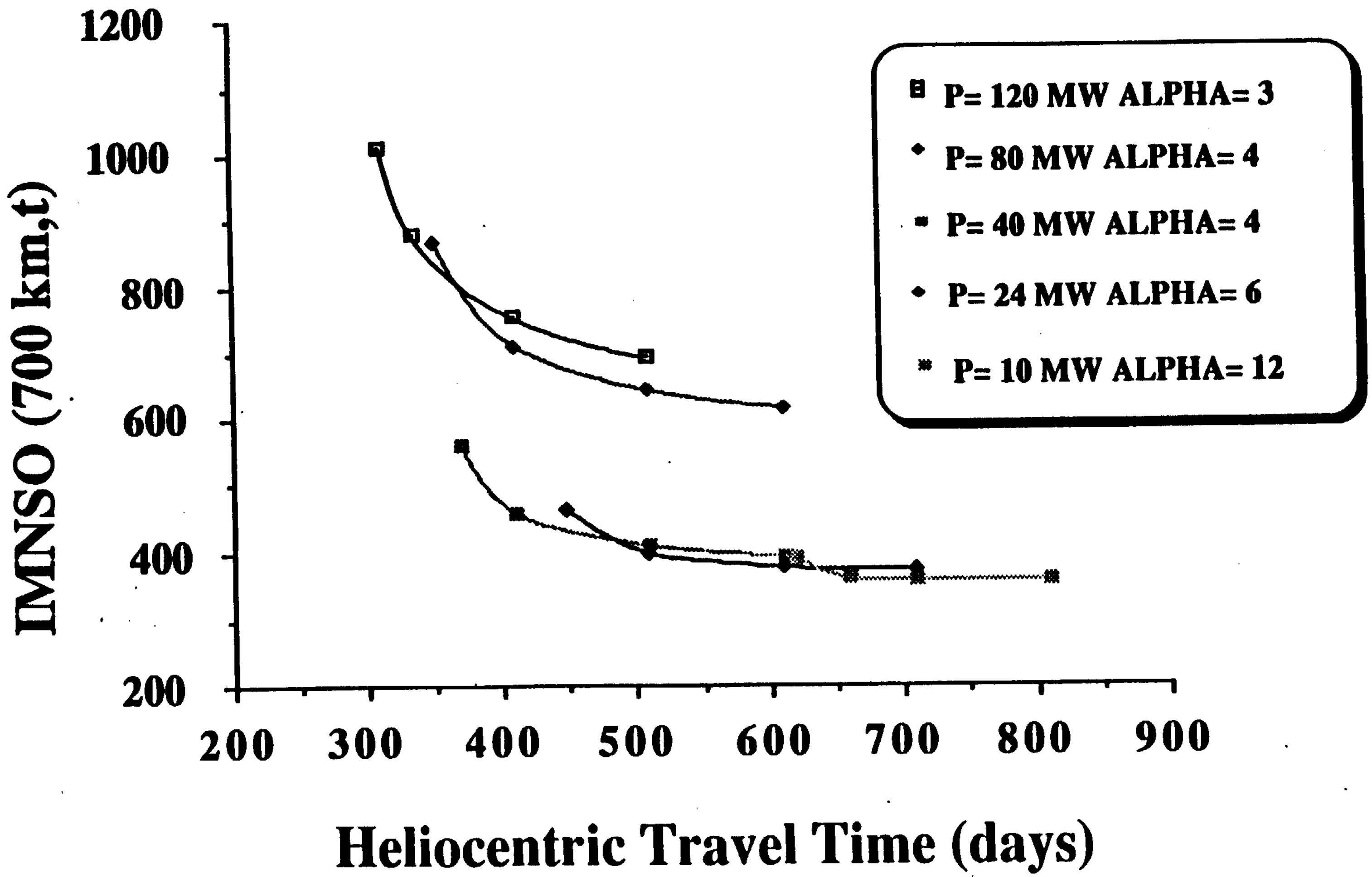
STCAEM/brc/16Mar90



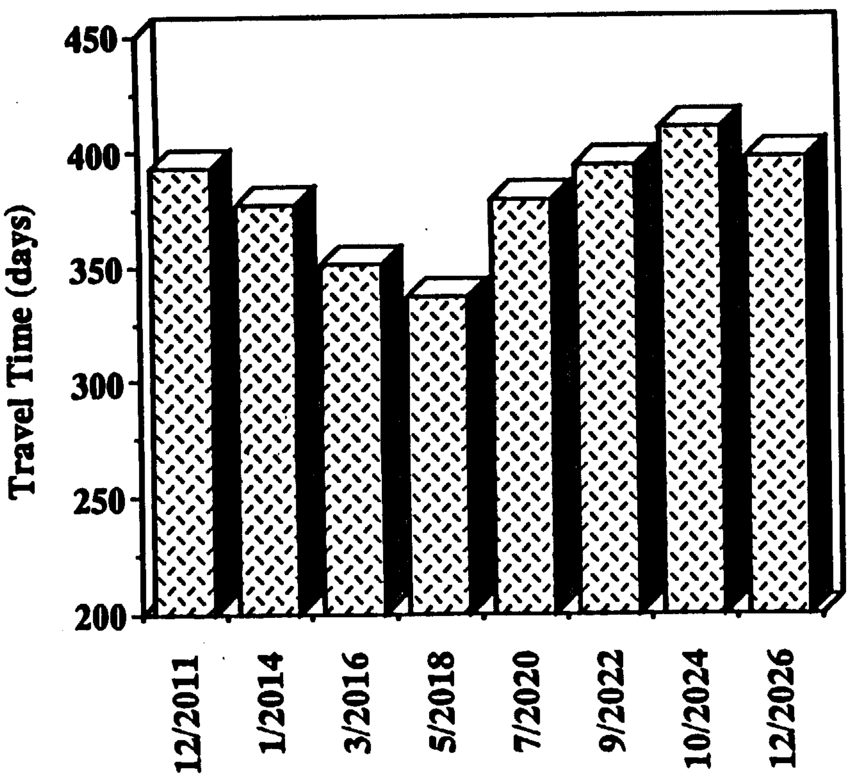
NEP Opposition Class Mission Opportunities

Mission analysis of the different opposition class missions reveals the optimum departure dates for a 40 MWe NEP vehicle with an alpha of 4 kg/kW. For these different opportunities, the minimum achievable heliocentric travel time is shown as well as the associated initial mass in Earth orbit. Results show that years 2016 and 2018 offer the shorter trip times in this cycle, while years 2020 and 2024 offer the lower mass. The variation in the total cycle is negligible when compared to chemical propulsion or other means with a lower Isp. The high Isp of electric propulsion (5,000-10,000 sec) offers mission flexibility as well as other advantages.

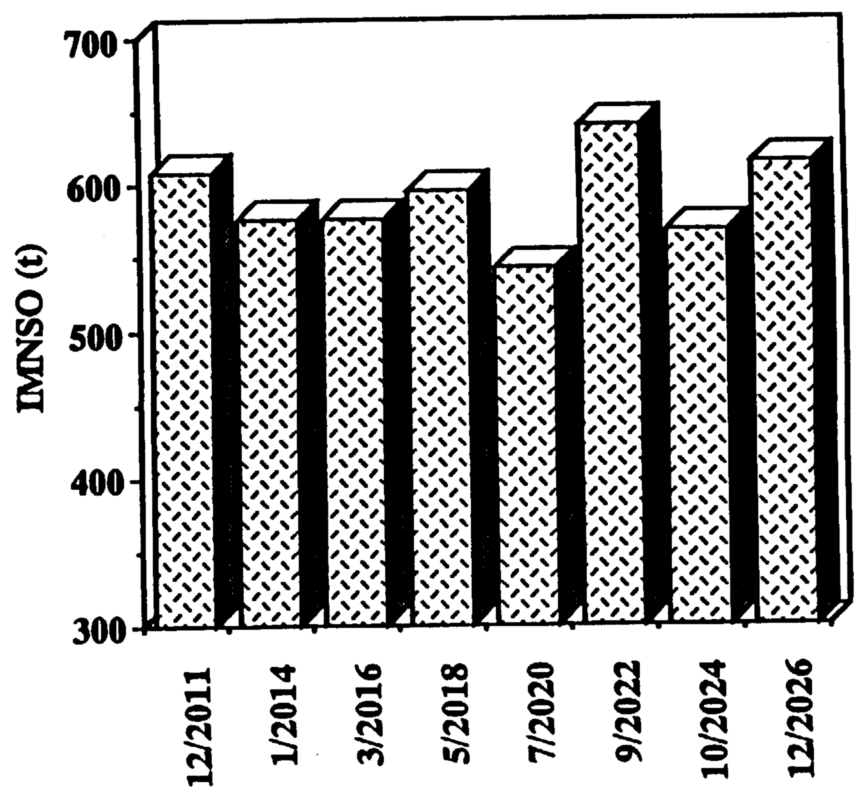
NEP Opposition Class Mission Opportunities

P= 40 MW Alpha= 4 kg/kW

Minimum Achievable Heliocentric Travel Time for The Different Opposition Class Missions



Associated Initial Mass in a 700 km Orbit (Nuclear Safe)



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ADVANCED CIVIL SPACE SYSTEMS

Summary of Space Nuclear Power Systems Launched by the United States (1961-84)

BOEING

STCAEM/brc/20Mar90

Power Source	Spacecraft	Mission Type	Launch Date	Status
SNAP-3B	TRANSIT 4A	Navigational	June 29, 1961	Successfully achieved orbit
SNAP-3B	TRANSIT 4B	Navigational	November 15, 1961	Successfully achieved orbit
SNAP-9A	TRANSIT-5BN-1	Navigational	September 28, 1963	Successfully achieved orbit
SNAP-9A	TRANSIT-5BN-2	Navigational	December 5, 1963	Successfully achieved orbit
SNAP-9A	TRANSIT-5BN-3	Navigational	April 21, 1964	Mission aborted: burned-reentry
SNAP-10A *	SNAPSHOT	Experimental	April 3, 1965	Successfully achieved orbit
SNAP-19B2	NIMBUS-B-1	Meteorological	May 18, 1968	Mission aborted: source retrieved
SNAP-19B3	NIMBUS III	Meteorological	April 14, 1969	Successfully achieved orbit
SNAP-27	APOLLO 12	Lunar	November 14, 1969	Successfully placed lunar surface
SNAP-27	APOLLO 13	Lunar	April 11, 1970	Mission aborted on way to moon, heat source returned to Ocean.
SNAP-27	APOLLO 14	Lunar	January 31, 1971	Successfully placed lunar surface
SNAP-27	APOLLO 15	Lunar	July 26, 1971	Successfully placed lunar surface
SNAP-19	PIONEER 10	Planetary	March 2, 1972	Successfully operated to Jupiter and beyond
SNAP-27	APOLLO 16	Lunar	April 16, 1972	Successfully placed lunar surface
TRANSIT-RTG	"TRANSIT" (TRIAD-01-1X)	Navigational	September 2, 1972	Successfully achieved orbit
SNAP-27	APOLLO 17	Lunar	December 7, 1972	Successfully placed lunar surface
SNAP-19	PIONEER 11	Planetary	April 5, 1973	Successfully operated to Jupiter, Saturn, and beyond
SNAP-19	VIKING 1	Mars	August 20, 1975	Successfully landed on Mars
SNAP-19	VIKING 2	Mars	September 9, 1975	Successfully landed on Mars
MHW	LES 8/9	Communications	March 14, 1976	Successfully achieved orbit
MHW	VOYAGER 2	Planetary	August 20, 1977	Successfully operated to Jupiter and Saturn
MHW	VOYAGER 1	Planetary	September 5, 1977	Successfully operated to Jupiter and Saturn

* Reactor

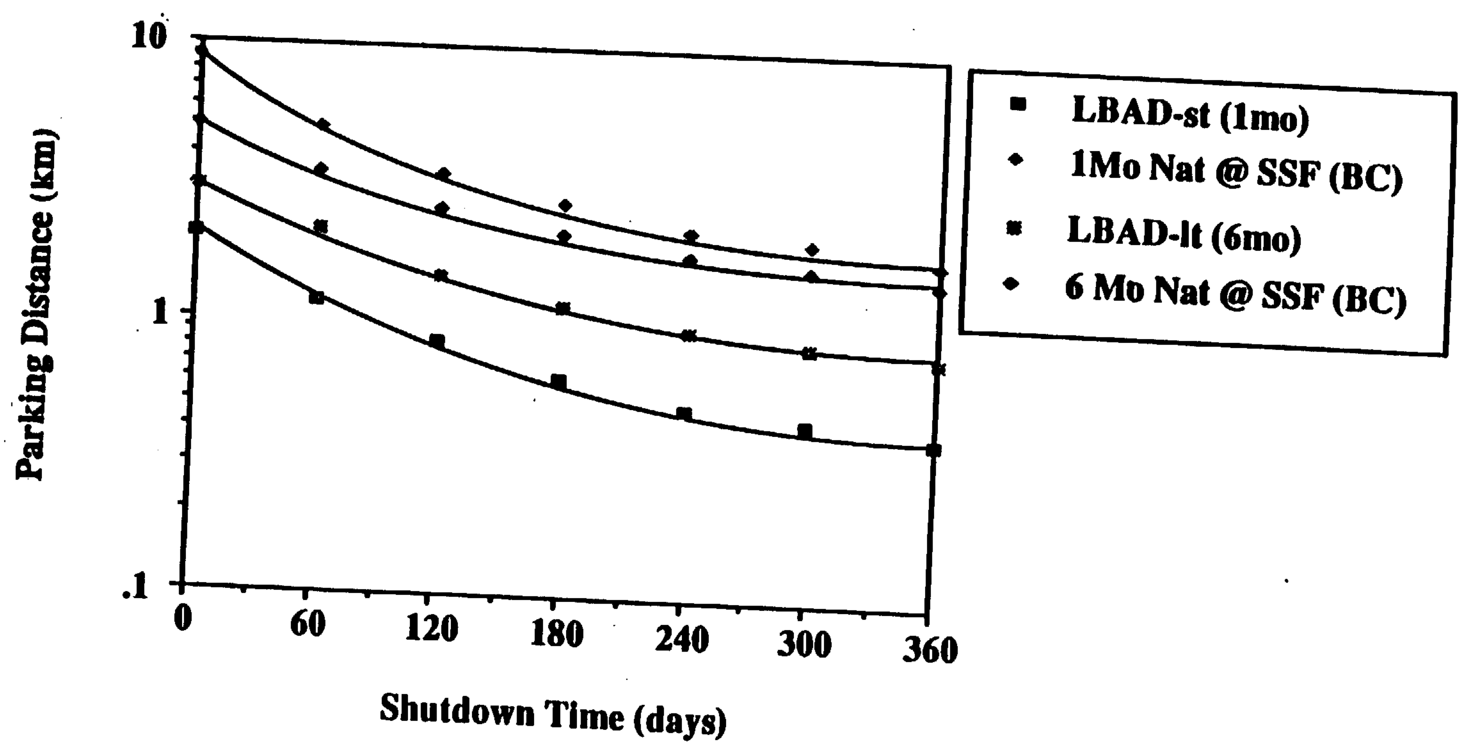
Nuclear Safety Operations

The following graph illustrates the 4-hour integrated dose equivalent that an EVA astronaut would receive outside the shadow shield at either 50, 100, or 200 meters from an NEP reactor after that reactor had been previously shutdown for times shown on the abscissa. For comparison, the one month limit to blood forming organs - short time and one month natural doses at SSF under worst-case (WC) and best-case (BC) conditions are shown. The information on this plot may be understood as follows. Assume the reactor had been shutdown for 150 days. Without any additional shielding, a 4-hour EVA could be performed up to 50 meters from the reactor before exceeding the short-term dose budget. If the EVA were performed at 100 meters from the reactor, the integrated dose would be reduced to a level equaling a one month natural exposure under worst-case conditions.

C 5

Nuclear Safety Operations

Parking Distances and Shutdown Times Required to Reduce Doses to Specified Levels for the NEP Reactor Outside the Shadow Shield



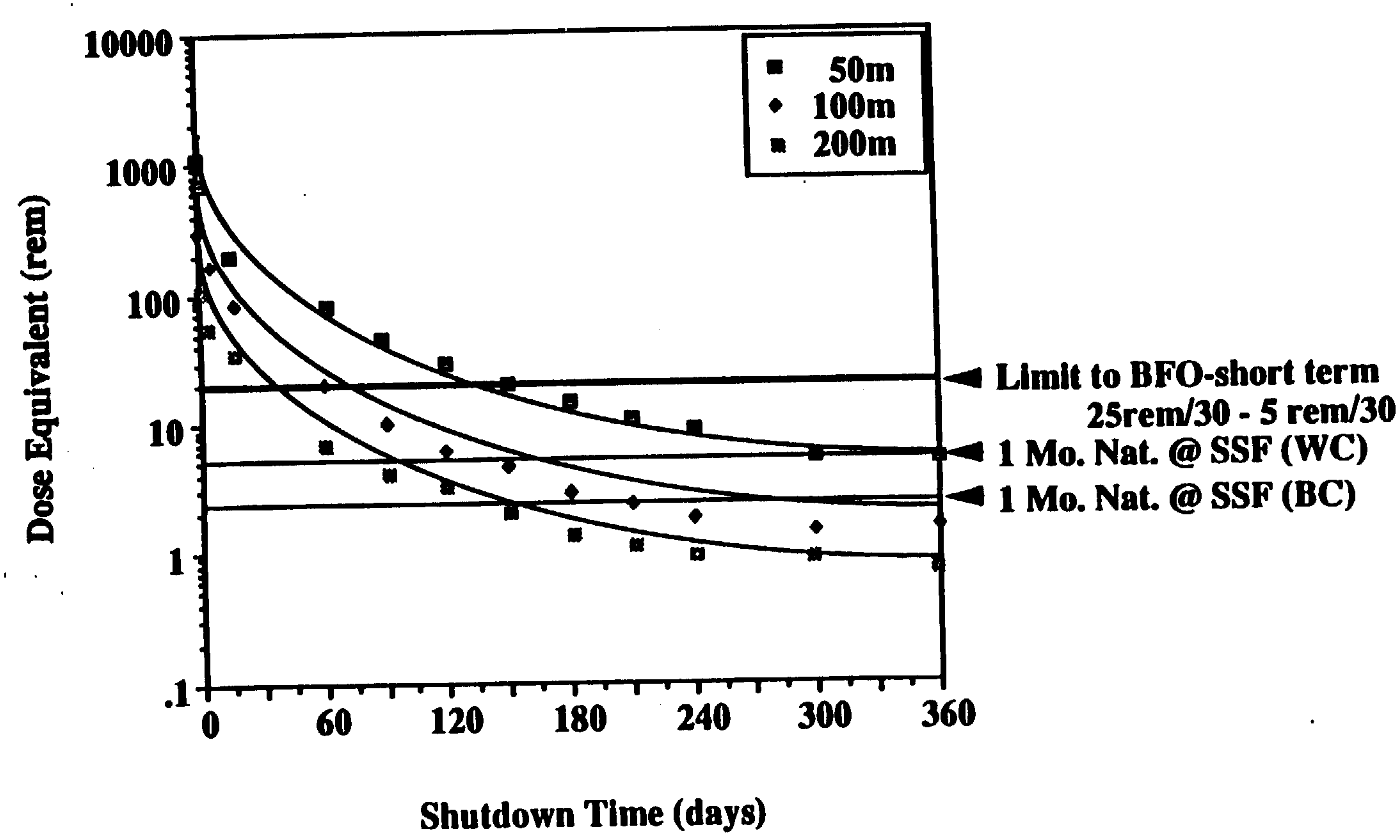
* Graph redrawn from "Radiological Assessment of Space Nuclear Power Operations from Space Station Freedom", Wesley et.al.

Nuclear Safety Operations

Without considering additional shielding, radiation doses to crew members living at space station may be controlled by either increasing the final parking distance of the vehicles or by allowing for greater shutdown times before towing the vehicles to SSF vicinity. The following graph gives isodose contours at four levels of integrated radiation dose as a function of both parking distance and previous shutdown time.

Nuclear Safety Operations

Four-Hour EVA Dose After Shutdown of NEP Reactor
Outside Shadow Shield



* Graph redrawn from "Radiological Assessment of Space Nuclear Power Operations from Space Station Freedom", Wesley et.al.

Nuclear Safe Orbit Considerations

Shown on the right are the possible departure and parking nodes that have been considered for nuclear vehicles. Also shown are the considerations or driving factors that go into node selection for a nuclear vehicle. Some of the problems associated with a node are listed as well as some options. If safety issues can be addressed, a SSF altitude node would reduce the number of confronting issues.

Nuclear Safe Orbit Considerations

- Nuclear safe altitude customarily set at 800 km for 300 yr life.
- The driving factors associated in selecting a node are:
  1. Safety
  2. Debris Environment
  3. Radiation Environment
  4. Mass Penalties Associated with Chemical Boost Stage
  5. Differential Nodal Regression
- Orbit accessibility will be ~1/year for 800km and ~2-3/year for >5000km
- Options:
  1. Operate nuclear system from SSF orbit, or
  2. Operate nuclear system from high orbit, above
     (a) debris environment
     (b) high-radiation part of van Allen belt (>5000km)

A SSF altitude parking and departure orbit would significantly reduce the number of confronting issues.

Possible Departure and Parking Nodes

Safety?	SSF-400 km
Debris	700-800 km
	~ 1200 km
Mass	GEO
	L2

Increased NEP Reliability

Currently the NEP reactor is a single point failure in the power chain. Analyses may reveal that the reactor will be considered as primary structure from a reliability standpoint. However two smaller reactors could be incorporated into a scenario that would ensure safe crew and vehicle return if a reactor need be shutdown during the mission. The worst case scenario would be to loose a reactor while spiraled down at Mars. Two options available to ensure safe crew return are carry more propellant or decrease Mars stay time.

Increased NEP Reliability

Issue: Presently the NEP reactor is a single point failure link in the power chain.

Possible Solution: Use two or more smaller reactors to furnish the required power.

Assumptions:
- Based on 40 MWe NEP reference case
- Worst case, reactor-out at Mars
- 2-20 MW reactors

Option 1

Carry More Propellant

Approximately 10 t of additional Argon propellant required in LEO

Option 2

Decrease Mars Stay Time

Decrease Mars stay time from 30 days to approximately 25 days

Either option will provide safe return of crew.

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NEP Power System Assumptions

Data from LeRC ASAO/HEI Office

- Current multi-megawatt reactor research has been limited to pulsed power applications (SDI) not continuous power applications, such as NEP.
- Brayton power conversion systems are at a higher level of development, but potassium Rankine conversion systems provide approximately 50% mass reductions. Since neither technology is currently flight-ready, and due to the mass reductions, the Rankine was selected as the more promising candidate.
- The reference system is composed of an SP-100 type lithium-cooled fast reactor in conjunction with multiple potassium Rankine power conversion loops.
- A shadow shield is used to provide a 5 rem/year dose rate at the payload, with 90% attributed to gamma rays and 10% to neutrons.
- The layered W/LiH shield provides for a 30m dia. dose plane at a separation distance of 100m.
- A minimum system mass was found to occur at a condenser temperature of 900K and an overall thermal-to-electric conversion efficiency of 20.8%.
- Results represent a power system mass using extrapolated SP-100 reactor technology and moderate technology development in fuels, materials, radiators, power conditioning, and potassium Rankine conversion systems.

10 MWe Man-Rated NEP Power System

A vehicle mass breakdown is provided for a 10 MWe NEP vehicle as well as technical specifications concerning the power source. The data on the power source was provided by the LeRC ASAO/HEI office. A total vehicle alpha of 11.8 kg/kW resulted from these calculations. It should be noted that a high amount of redundancy is included in these preliminary calculations.

10 MWe Man-Rated NEP Power System

STCAEM/brc/20Mar90

Data from LeRC ASAO/HEI Office

Reactor	Li-cooled Pin-type
Power Conversion	Potassium Rankine
Power Output	10 MWe
Full Power Life	10 yrs.
Turbine Inlet Temp	1300 K
Condenser Temp	900 K (min mass)
Thermal-Electric Eff.	20.8%
Reactor	
Coolant	Lithium
Fuel	UN pins
Cladding	PWC-11

Dose Constraint	5 rem/yr
Materials	W/LiH
Dose Plane Diameter	30m
Separation Distance	100m
Heat Rejection	
Type	Heat Pipe
Geometry	Planar
Specific Mass	5.5 kg/m^2
Total Radiator Area	1842m^2

System Mass Breakdown

Power System:

Reactor	11973 kg
Shield	14907 kg
Power Conversion	24247 kg
Radiators	10143 kg
Power Conditioning	17792 kg
Total	79063 kg

Specific Mass ... 7.9 kg/kW

Propulsion System and Other:

Thrusters	10000 kg
Structure	6000 kg
Communications	1500 kg
Avionics	500 kg
Experimental Platforms	1500 kg
Total	19500 kg

Vehicle Alpha- 10 MW

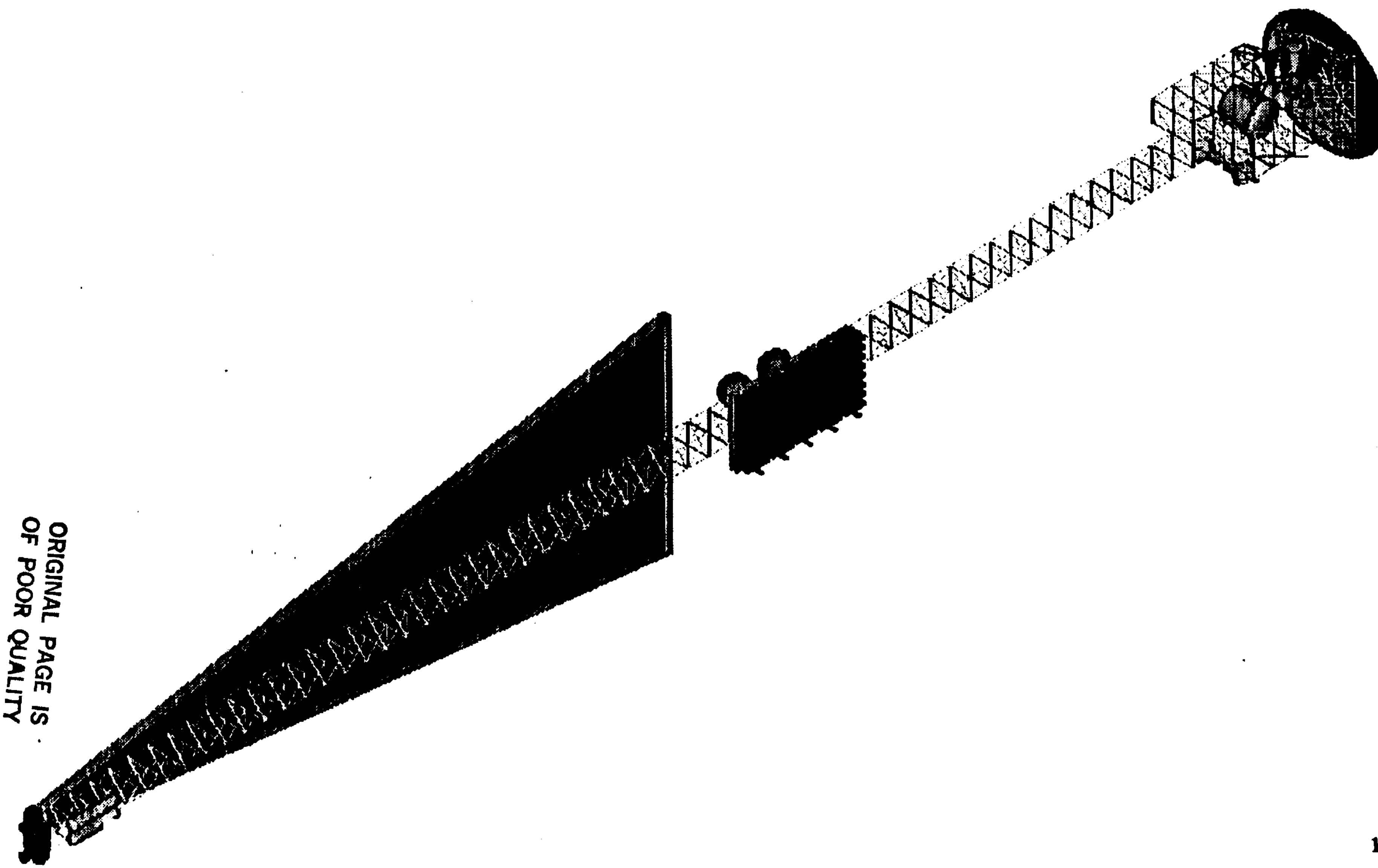
Total Vehicle Weight
98,563 kg

Vehicle + 20% Cont.
118,275 kg

Alpha = 11.8 kg/kW

NEP Configuration

The following charts depict the reference nuclear electric propulsion vehicle that has been modeled on the Intergraph CAD workstation. Many views are shown to provide the detail that the vehicle has been designed to. The vehicle model has verified conceptual design.



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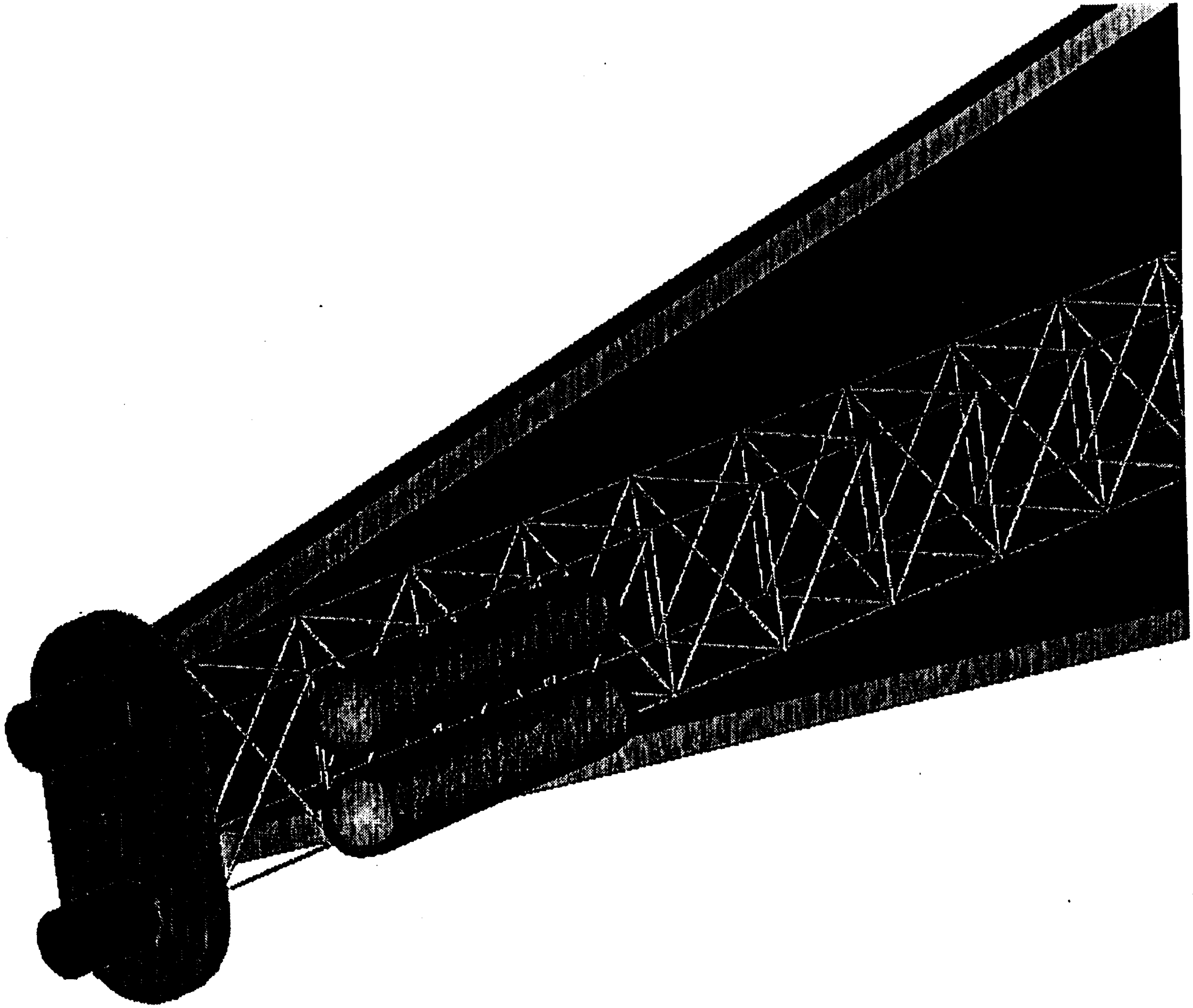

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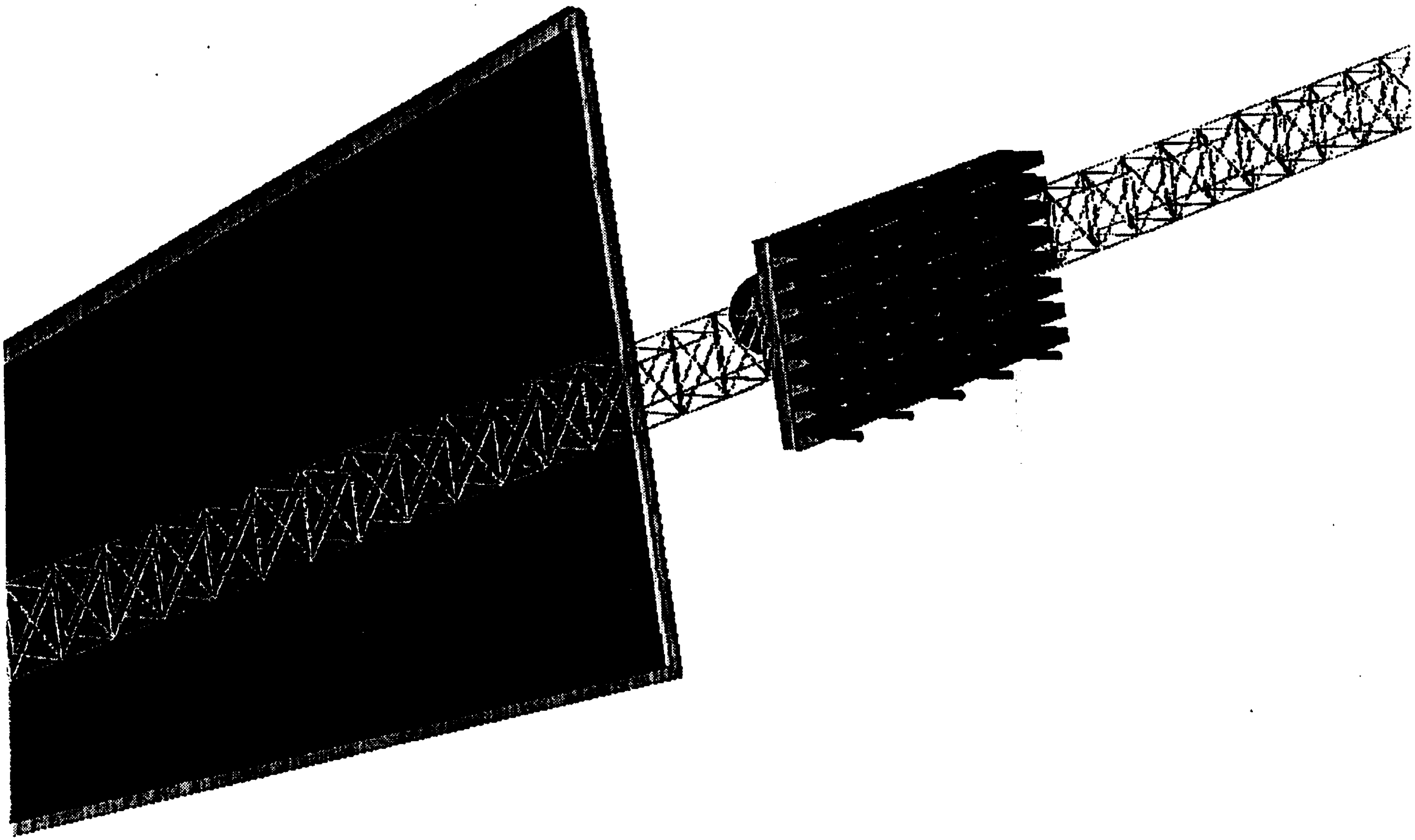

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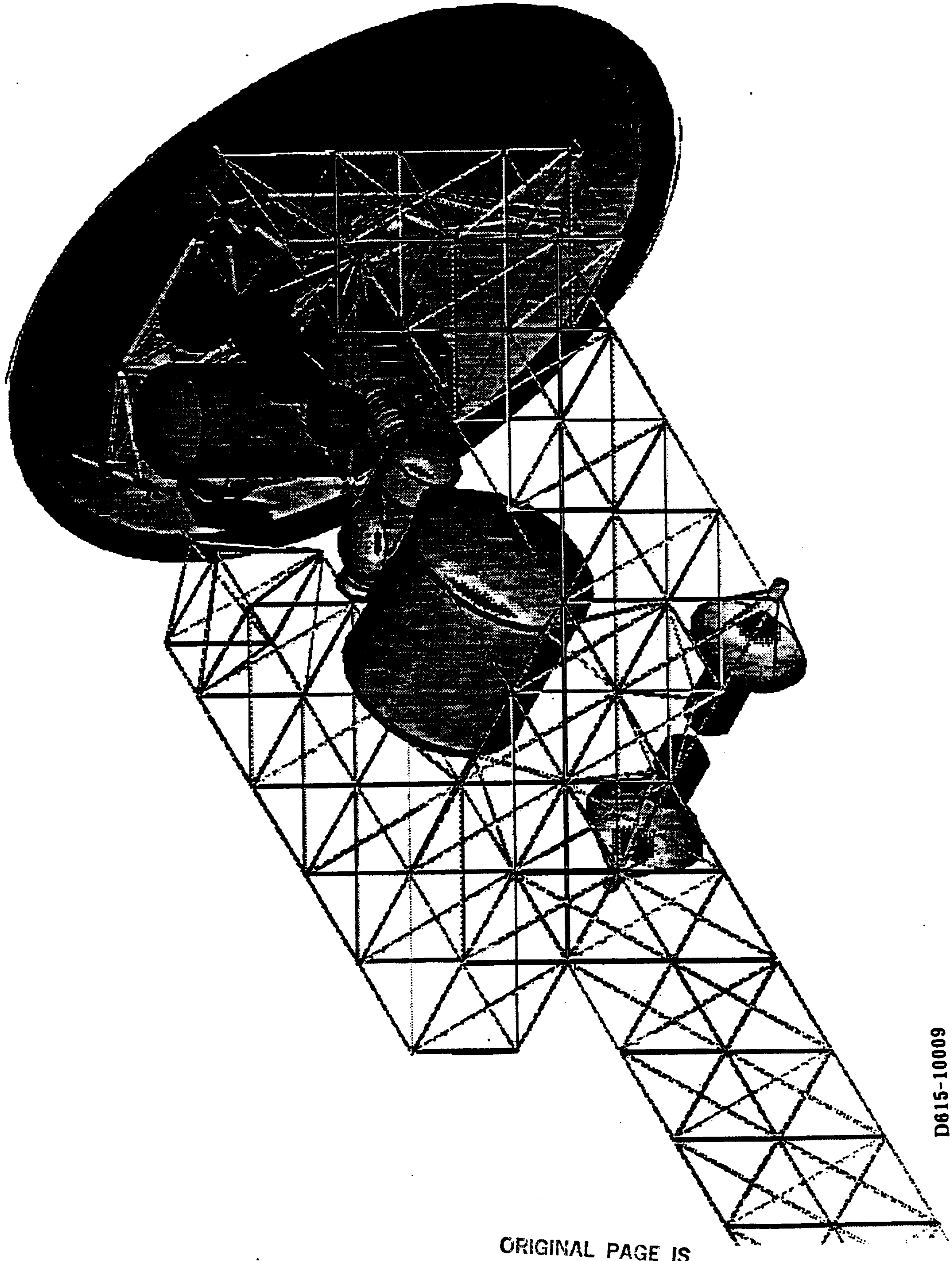

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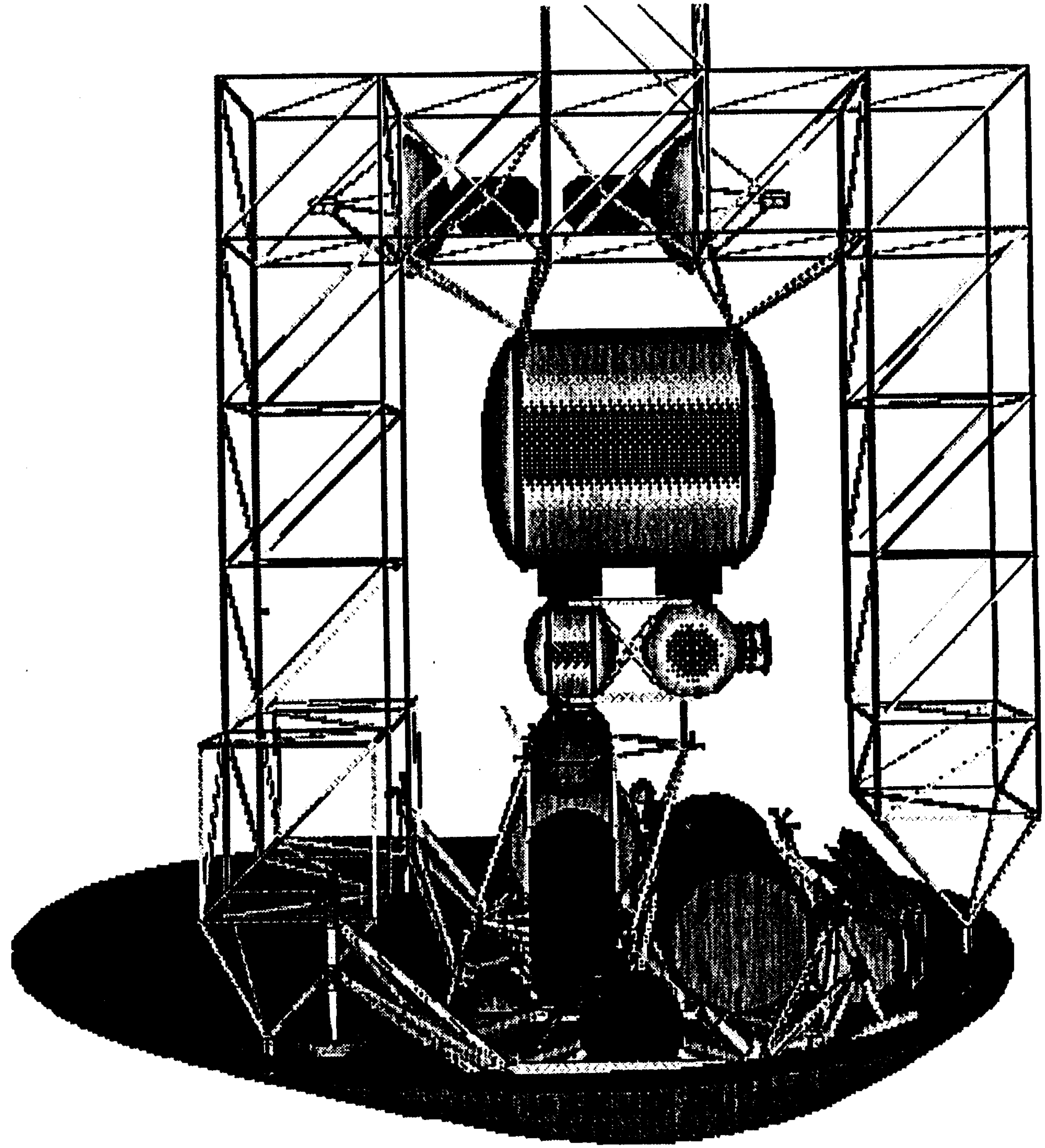

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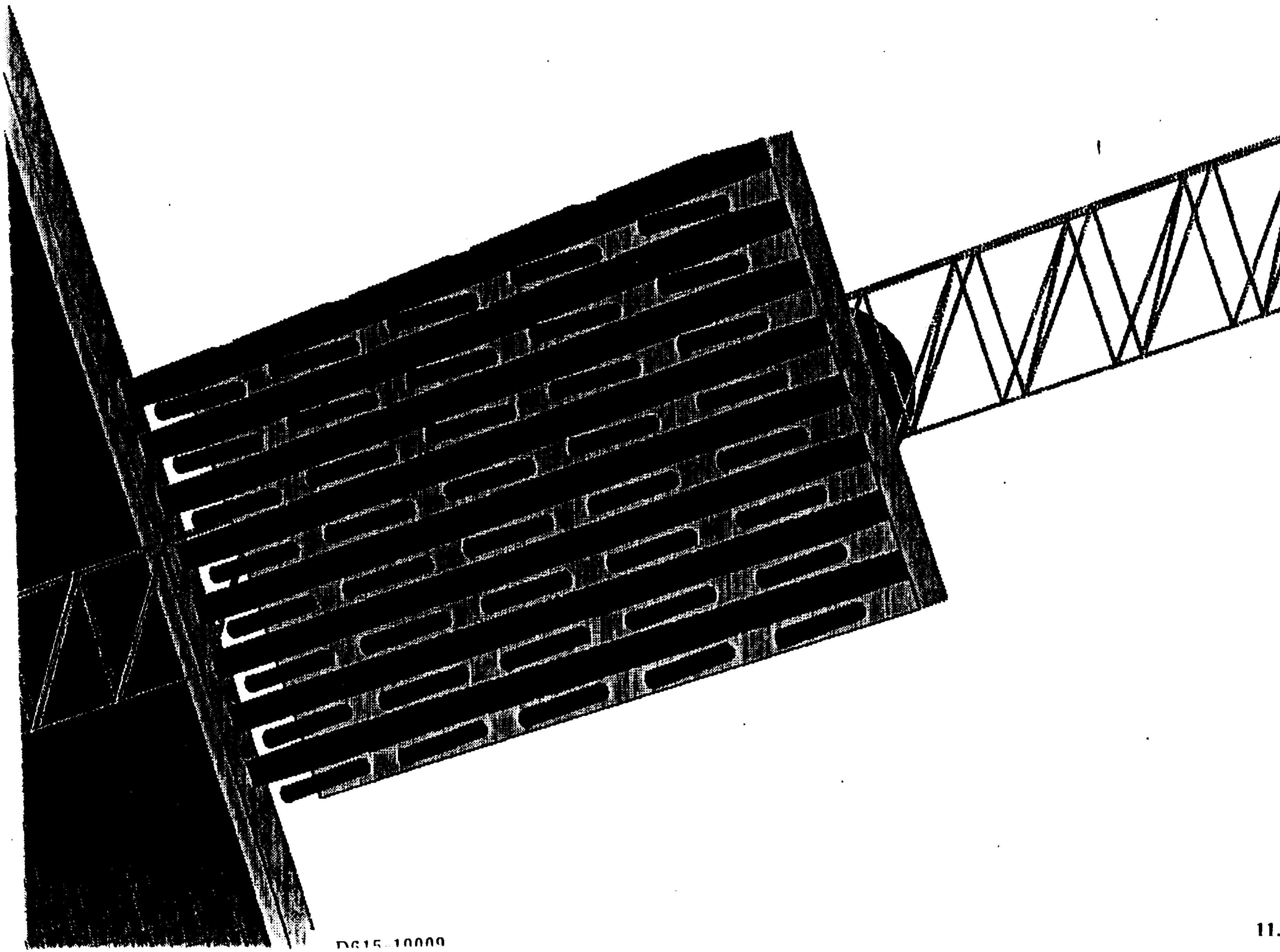

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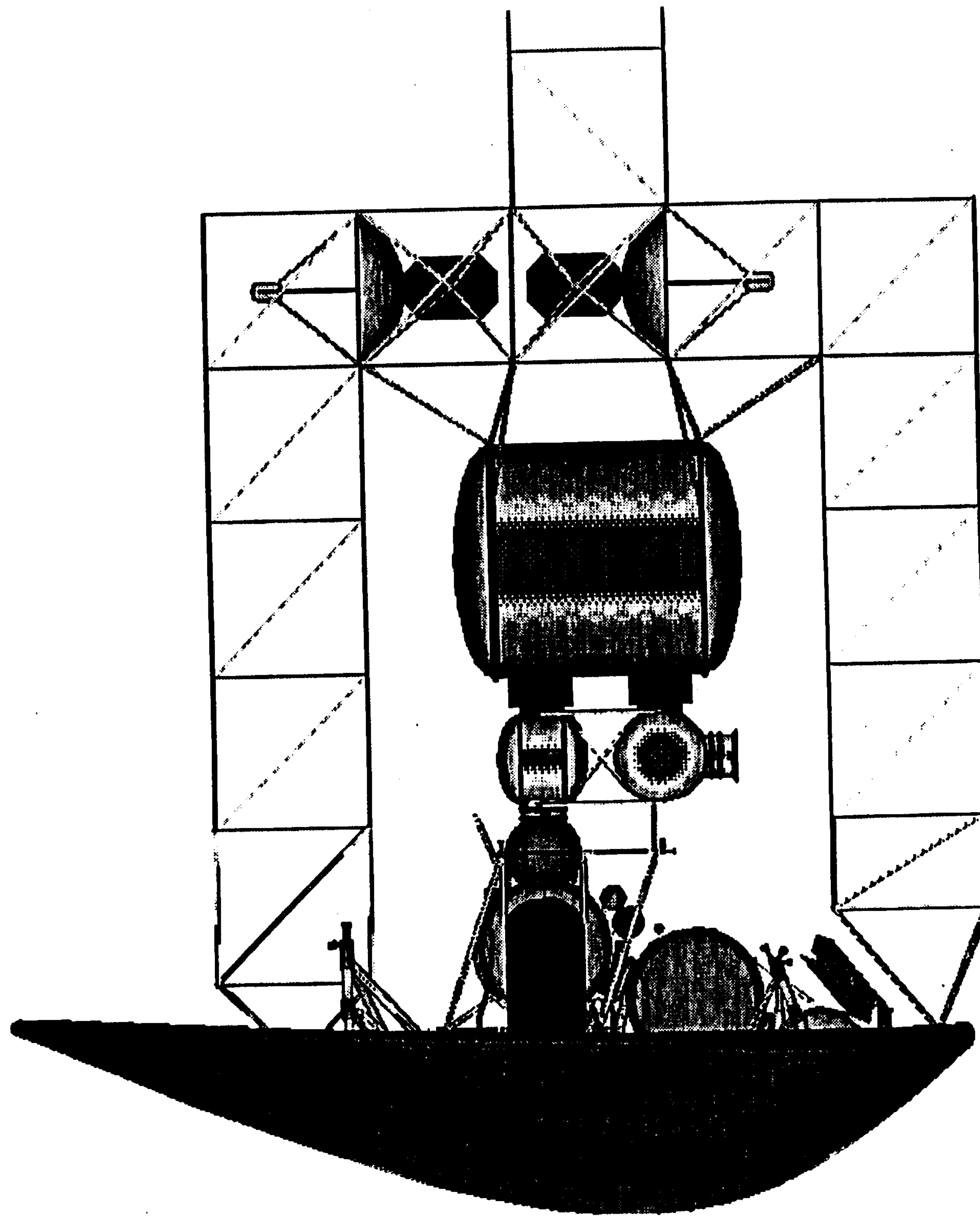

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Principal Findings- NEP

- A SSF altitude orbit will eliminate debris and mass penalties associated with higher orbits.

- Preliminary analysis shows that safety issues can be resolved for a SSF altitude parking and departure orbit.

- Main nuclear safety operations issue is Earth-to-orbit launch, not node selection.

- Years 2016 and 2018 offer best trip times.

- 2 smaller reactors increase reliability without significant penalties.

- A total vehicle alpha of ~12 kg/kW(includes 20% contingency) is reasonable to assume for a 10 MW NEP vehicle.

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SEP Update

Brad Cothran

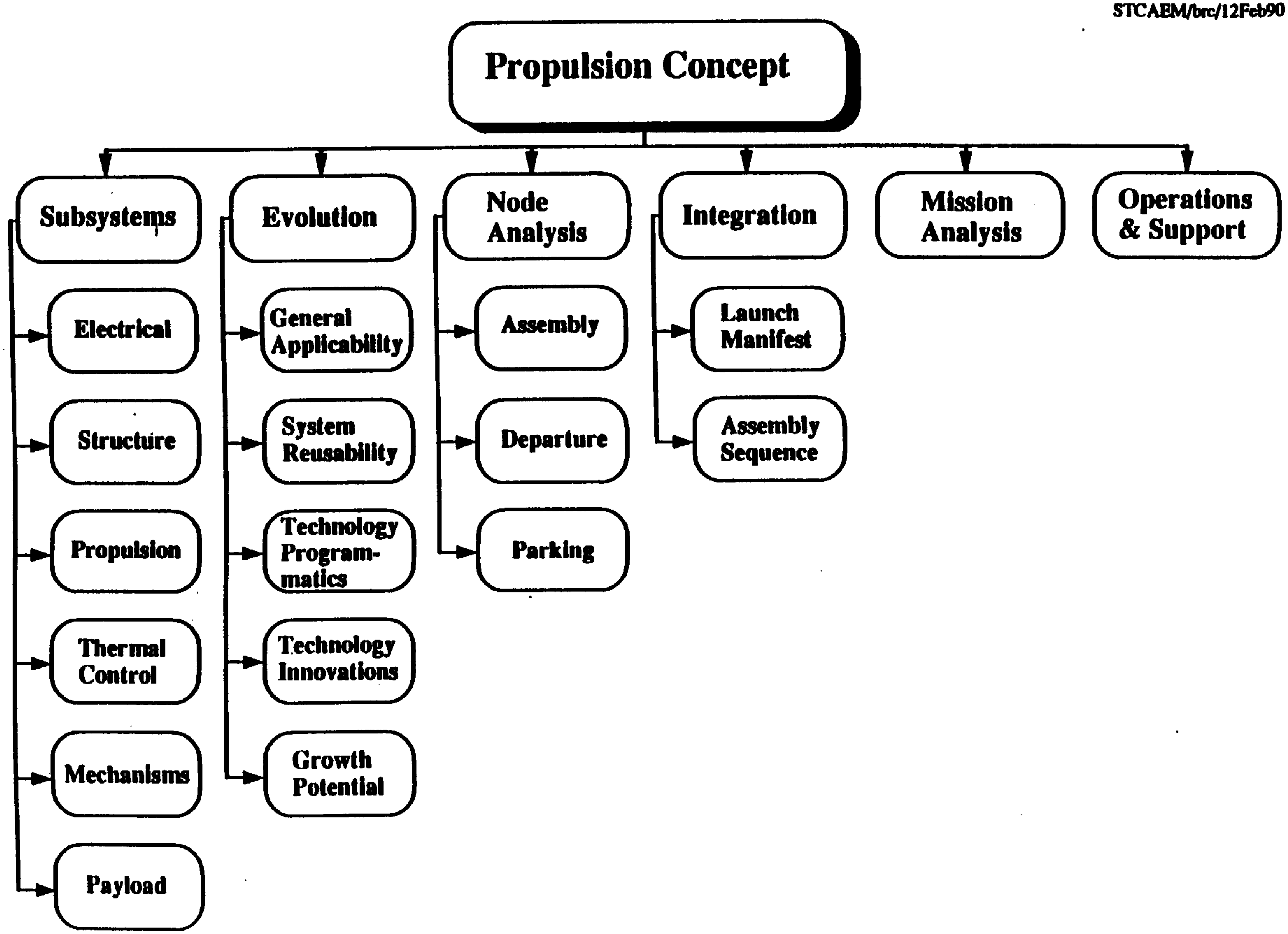
Agenda

Advanced Propulsion WBS
Configuration
Transfer Array
Flybys
Principle Findings

Advanced Propulsion WBS

Advanced propulsion options are shown and the method in which they feed into the contract. Other propulsion options are listed that are recognized, but are not given serious consideration due to technological feasibility or other reasons. Also shown is the breakdown of work to be performed on the main options under consideration. The structure allows for new ideas or concepts to be integrated into the overall system architecture.

Advanced Propulsion WBS (cont.)



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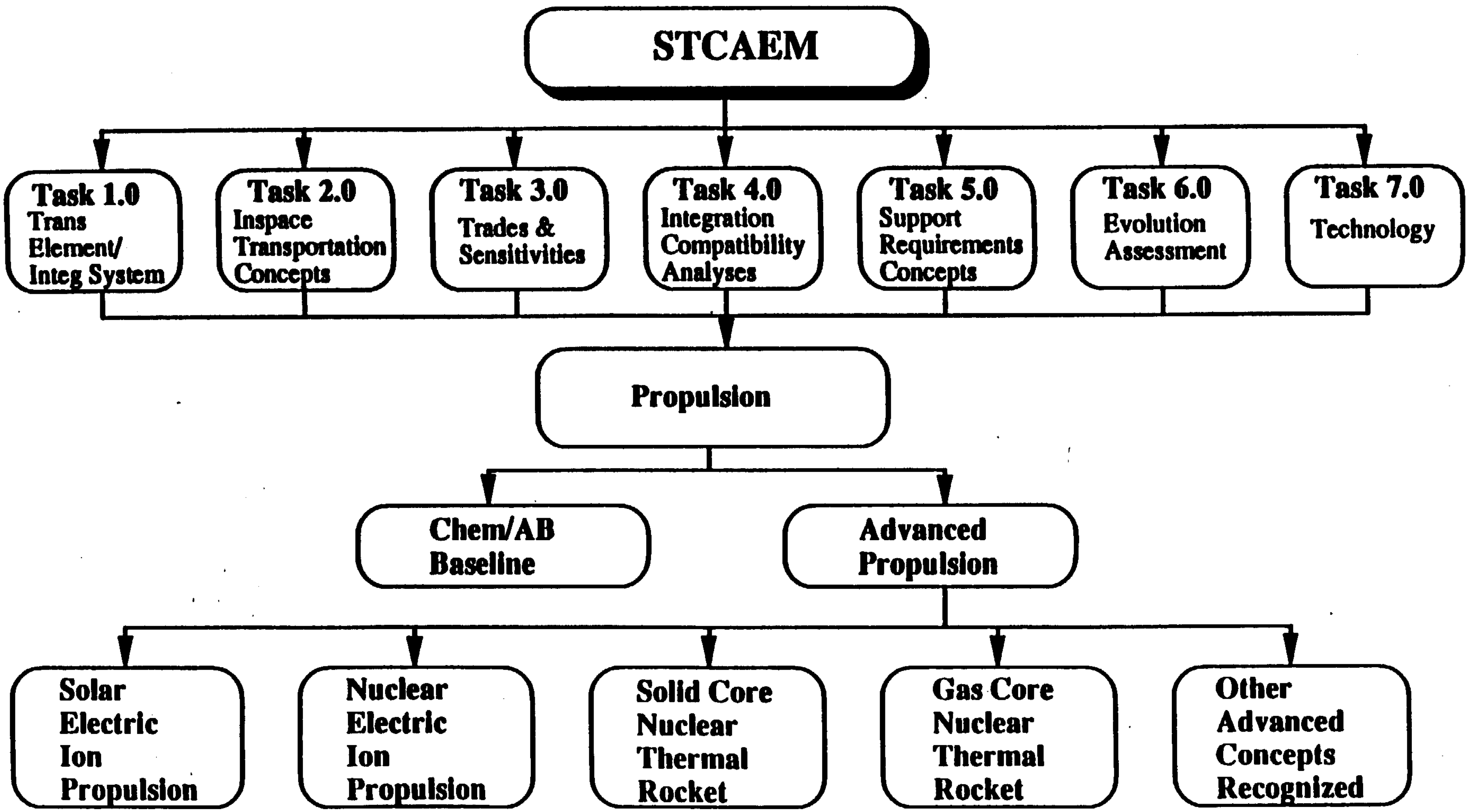
Advanced Propulsion WBS



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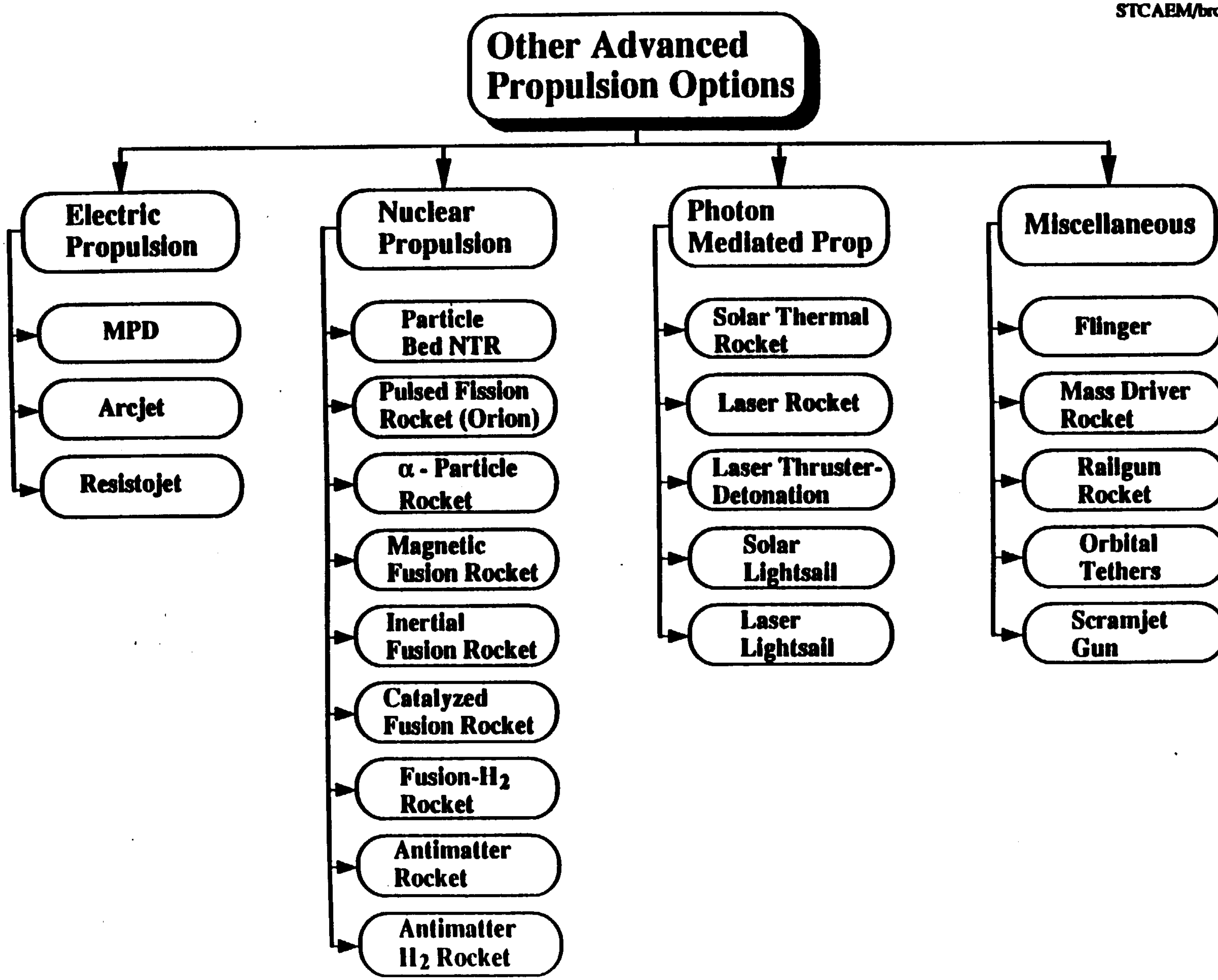
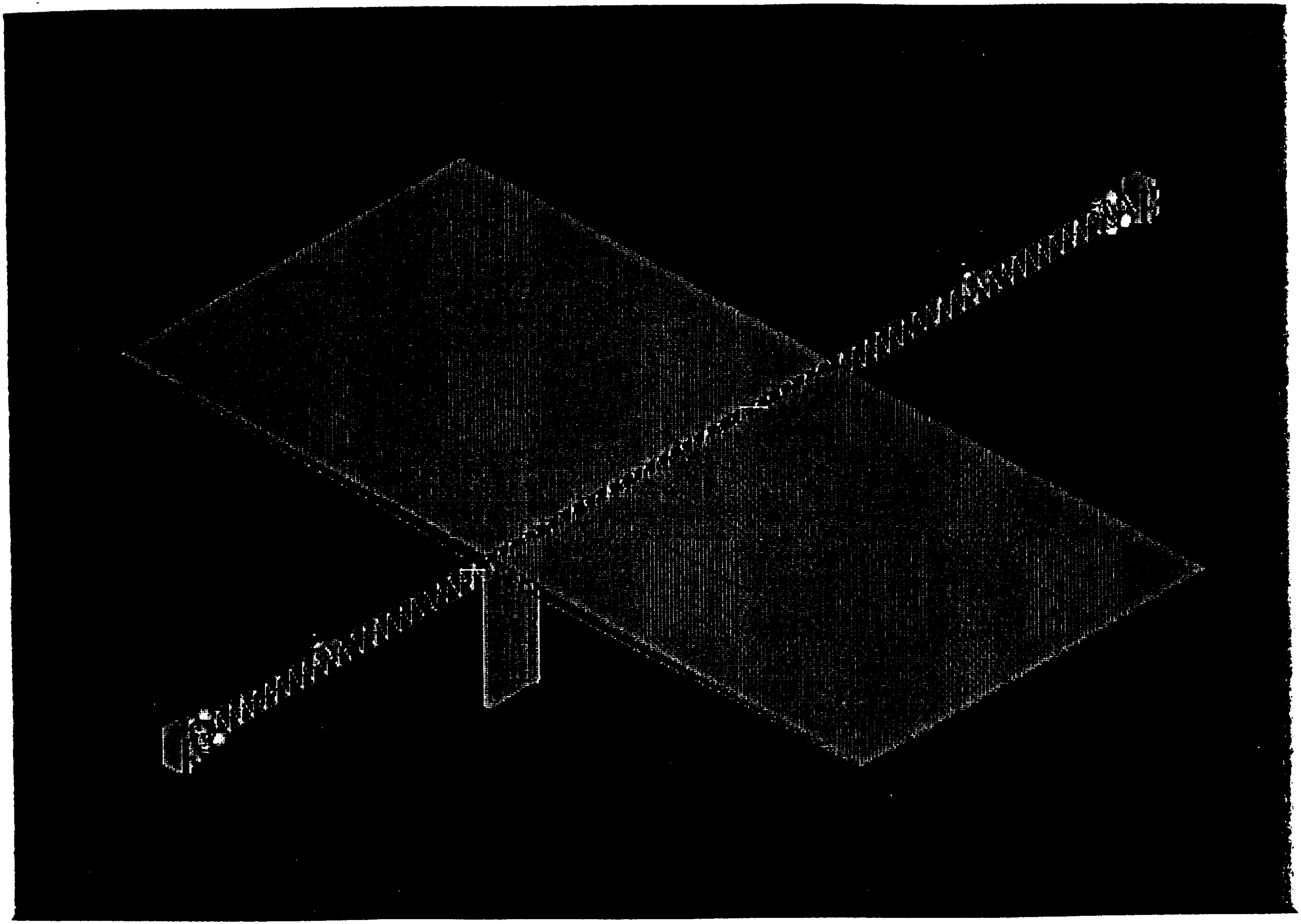
Advanced Propulsion WBS (cont.)



SEP Configuration

The following charts depict the reference solar electric propulsion vehicle that has been modeled on the Intergraph CAD workstation. Many views are shown to provide the detail that the vehicle has been designed to. The vehicle model has verified conceptual design.



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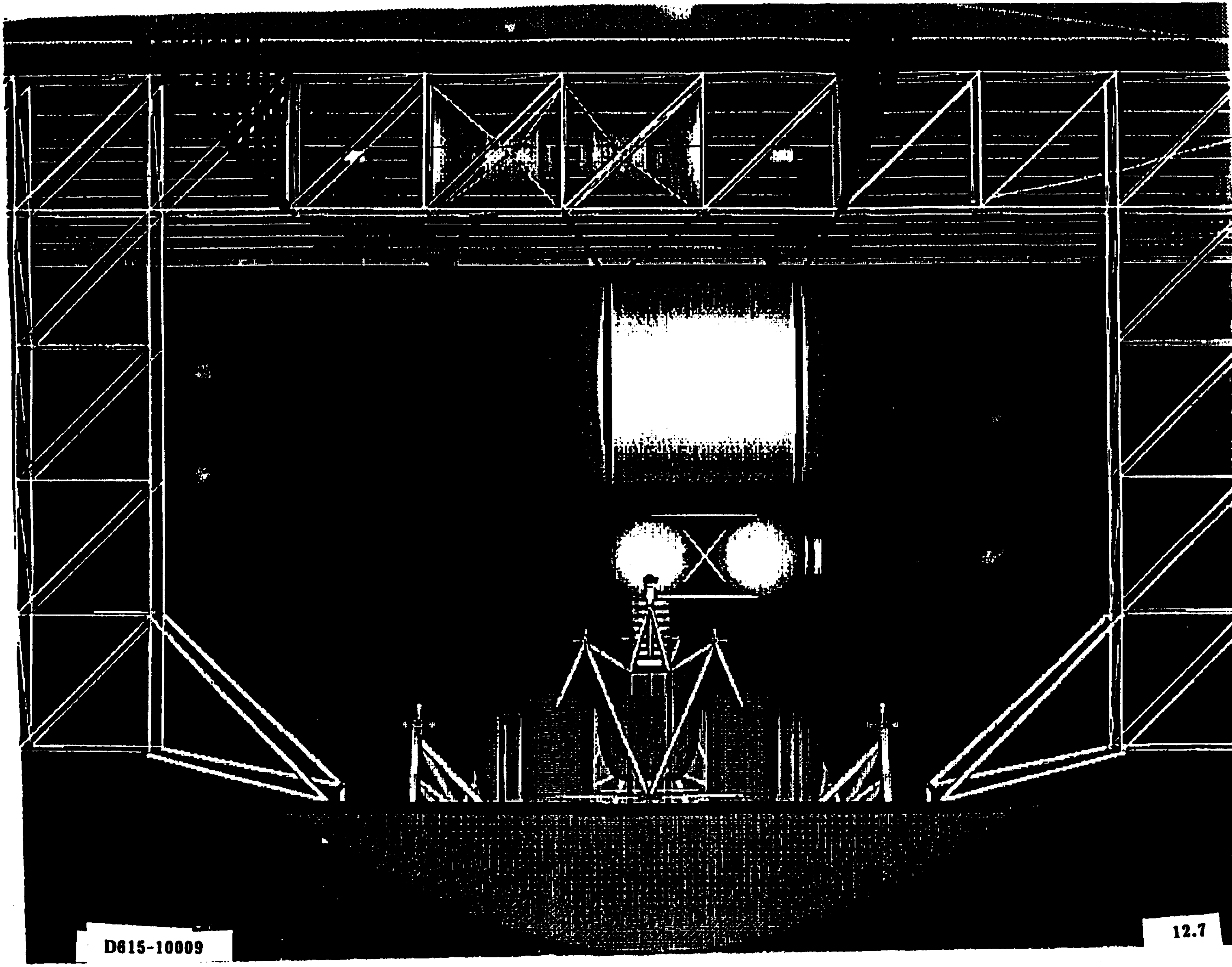
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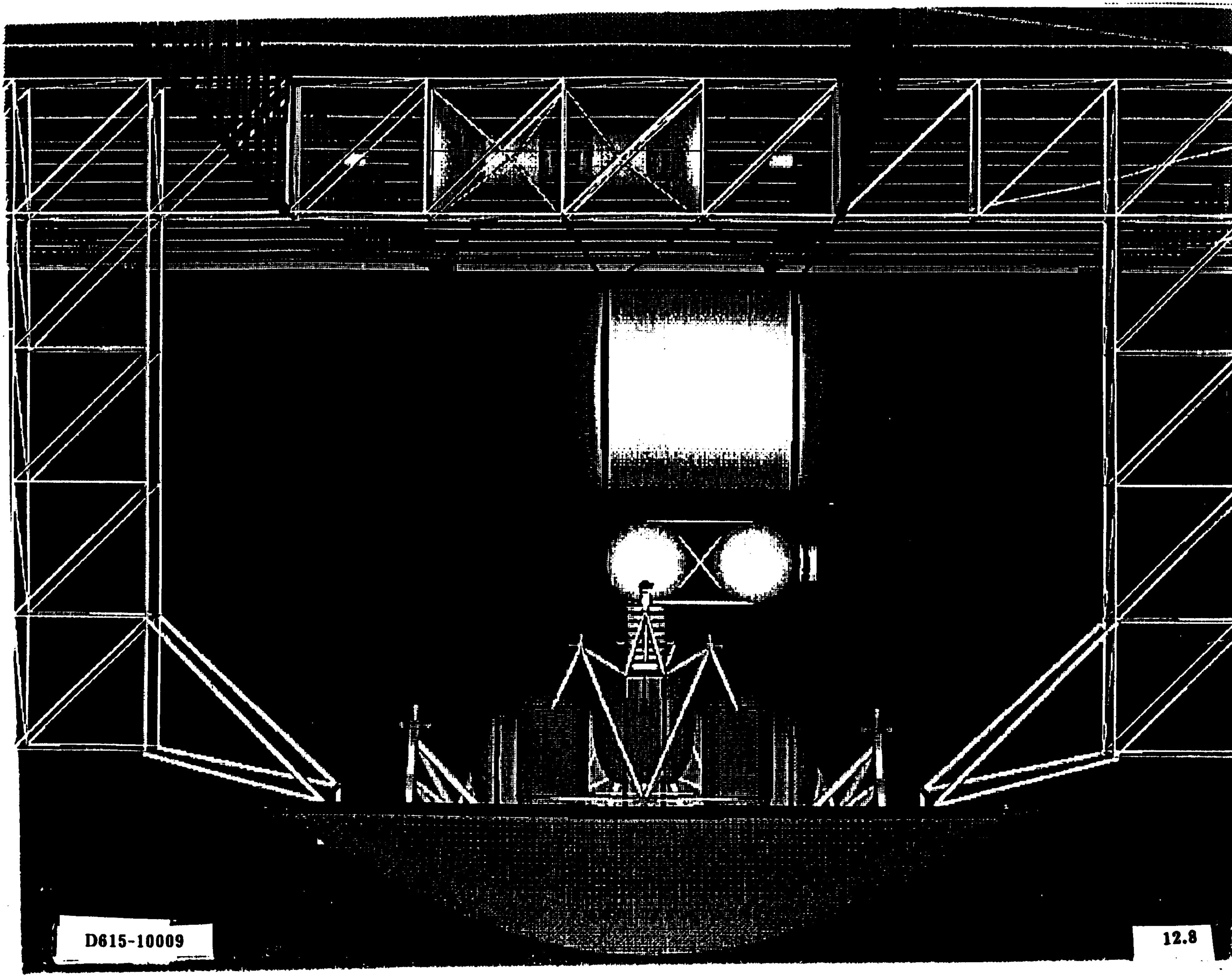
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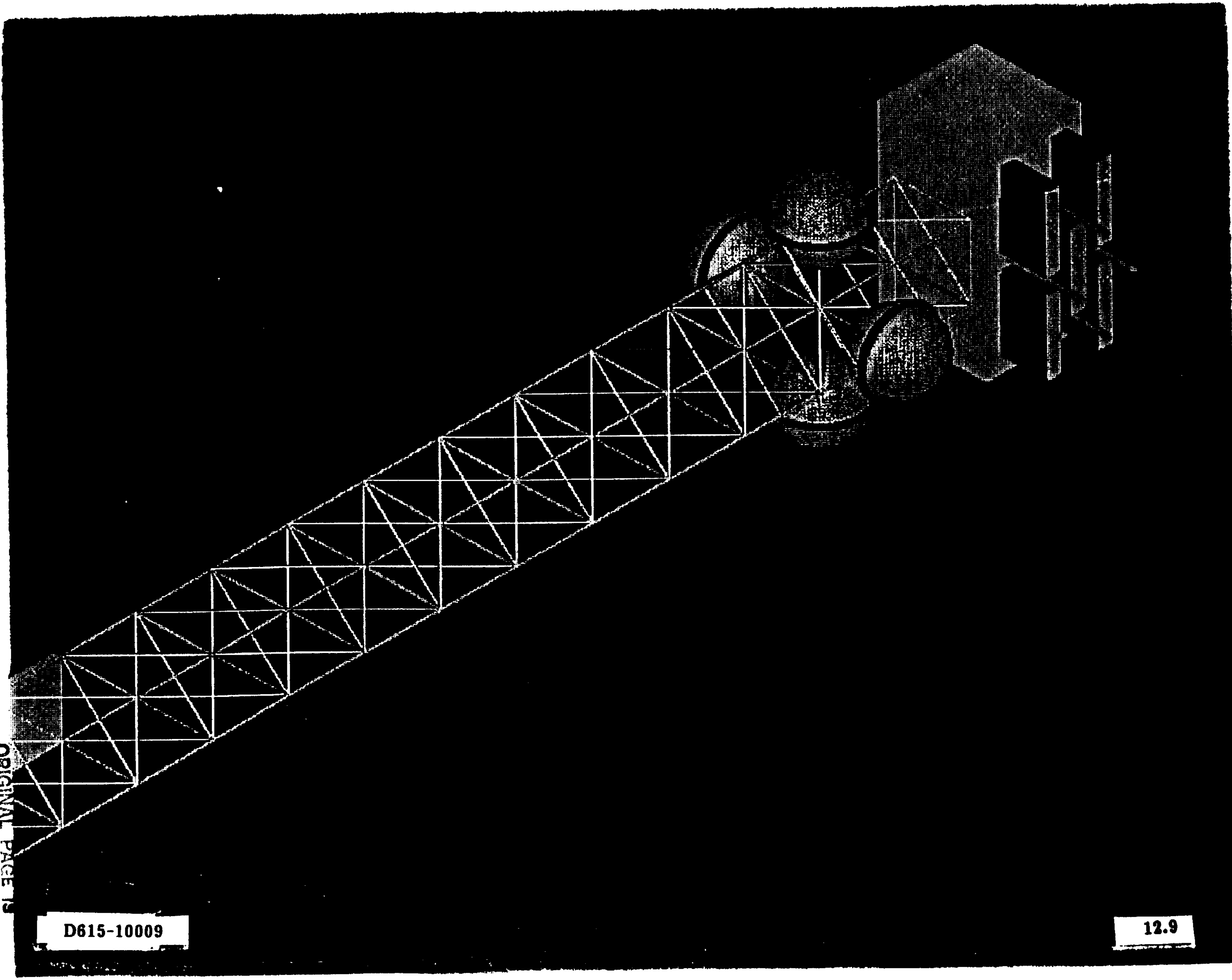

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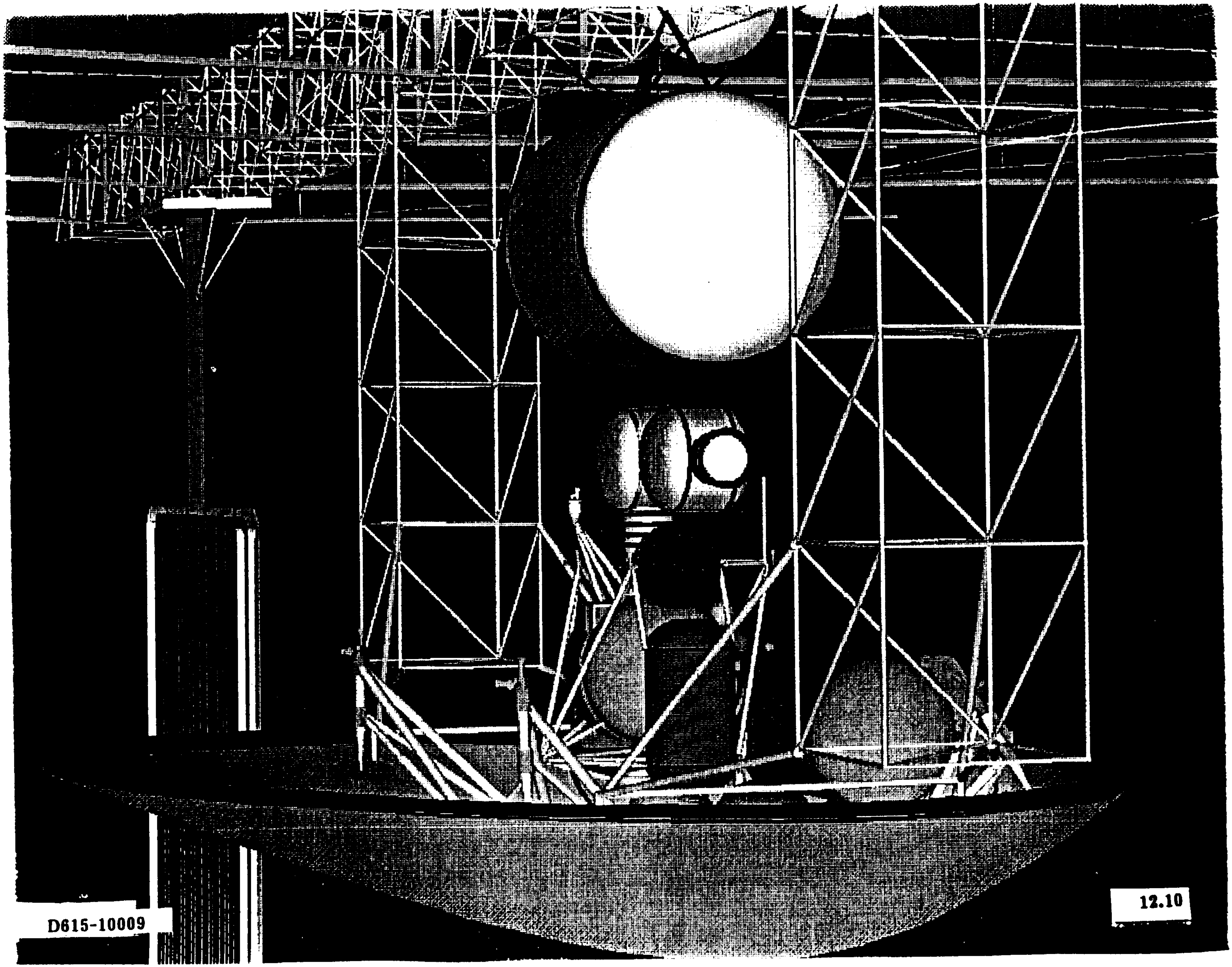
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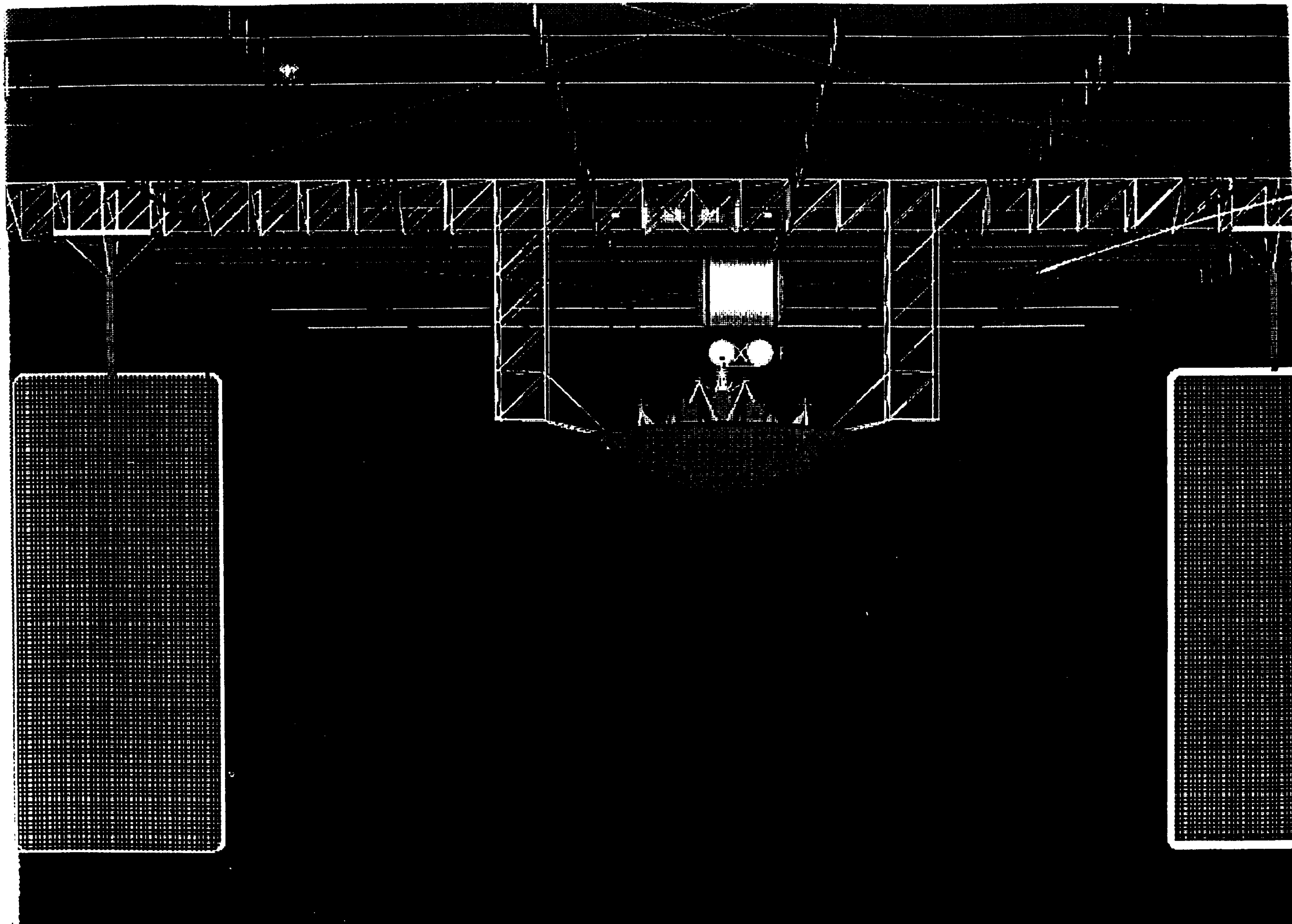

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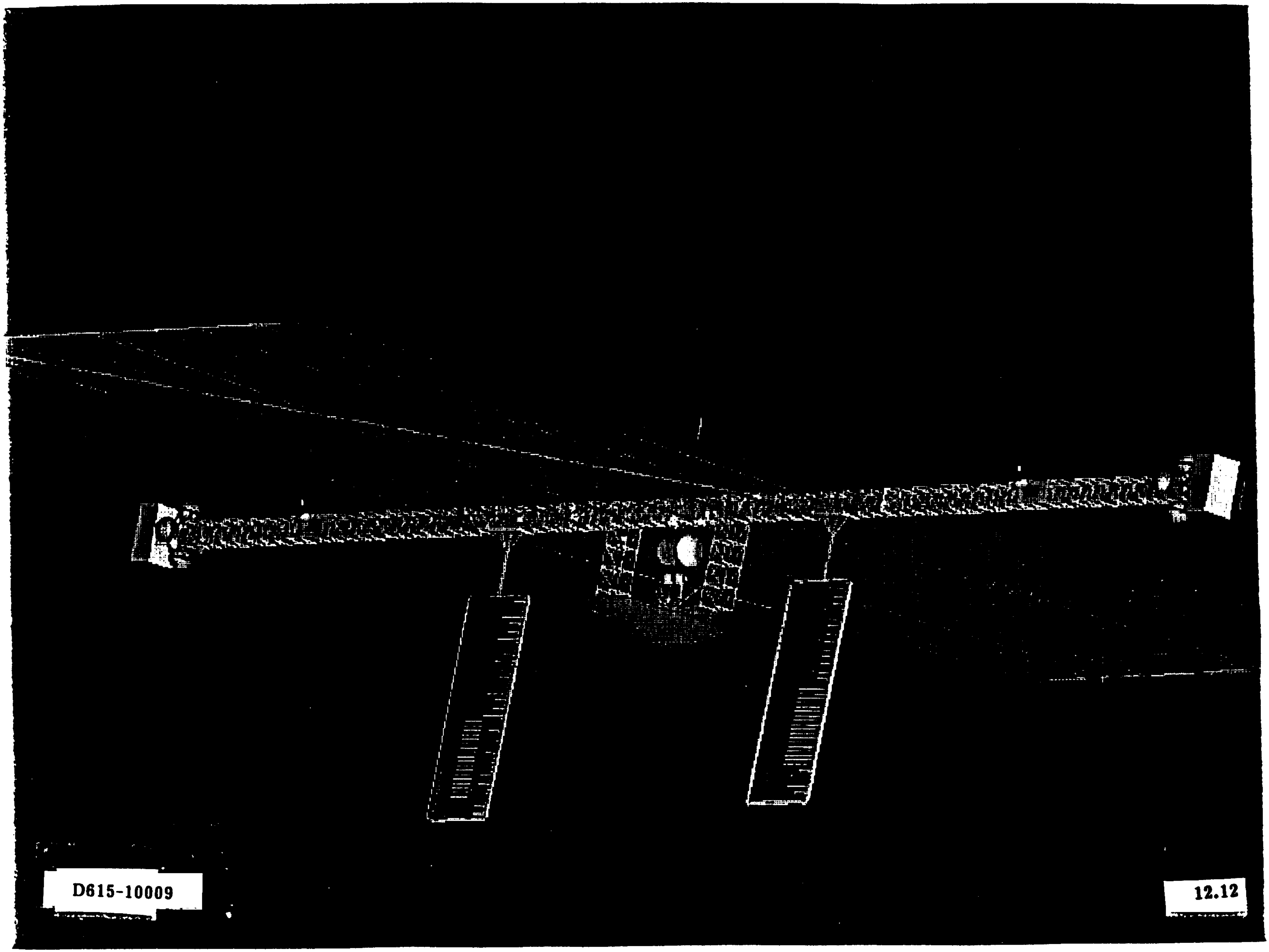
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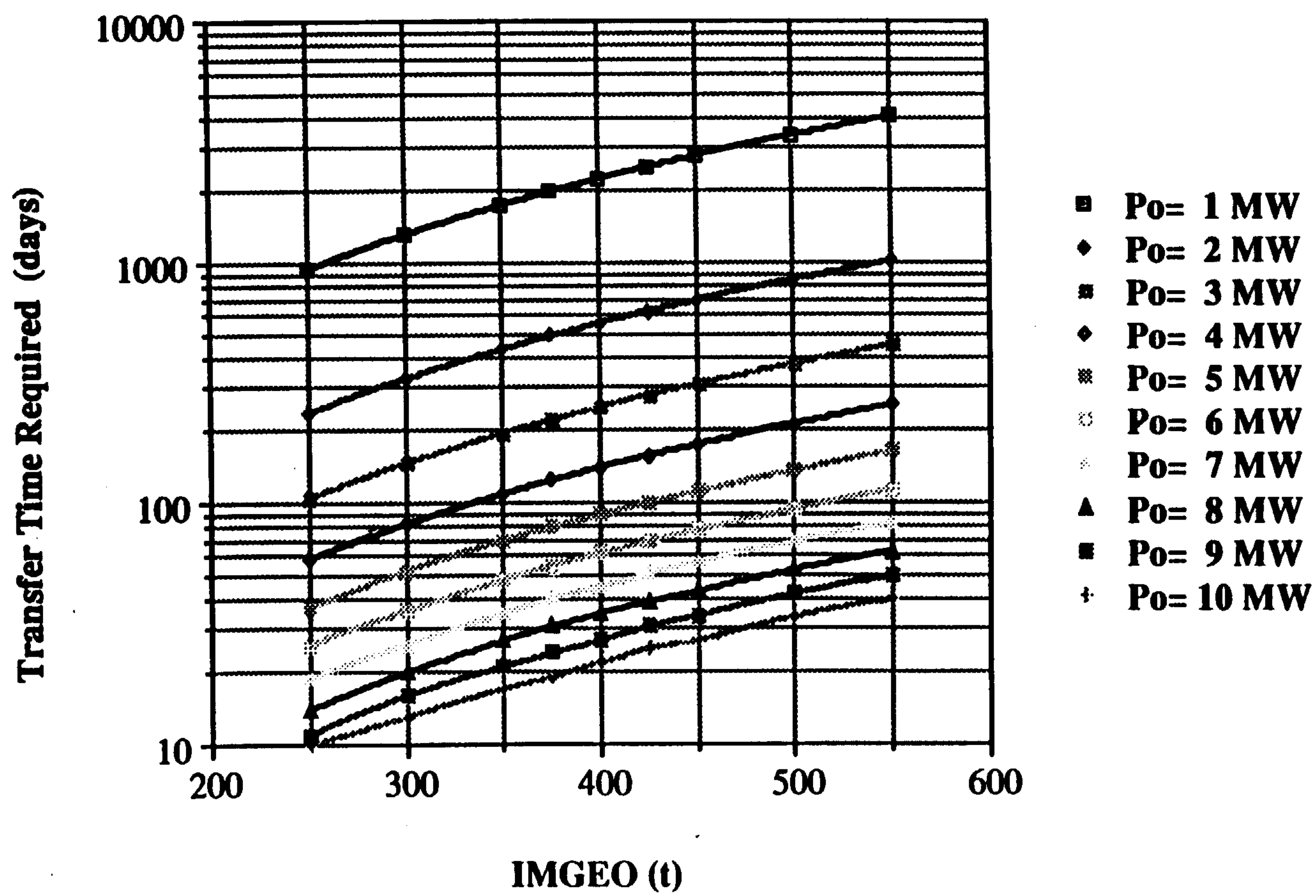
Spiral Time Analysis for Earth Spiral

Transfer time required in days is shown for corresponding initial masses at GEO for the different power levels. The referenced transfer time is the time it takes the SEP vehicle to spiral from LEO to GEO with its own propulsion system. This analysis was performed to determine the time penalties associated with the transfer array scenario (TAS). The transfer array scenario was developed to eliminate the heavy chemical boost stage necessary to transfer the vehicle from LEO to GEO. The TAS will provide a LEO mass savings benefit on the order of 200t. Once the SEP vehicle reaches GEO or a higher orbit, it can drop the arrays of to be used for a lunar power beaming platform or other uses. The arrays will experience roughly a 35% degradation due to the spiral through the van Allen belts. Some issues to be resolved are the amount of debris and radiation damage the vehicle will experience while traversing the belts. A power level of 5 MWe will transfer the vehicle in less than 100 days.

Spiral Time Analysis for Earth Spiral

LEO to GEO Electric Spiral Analysis

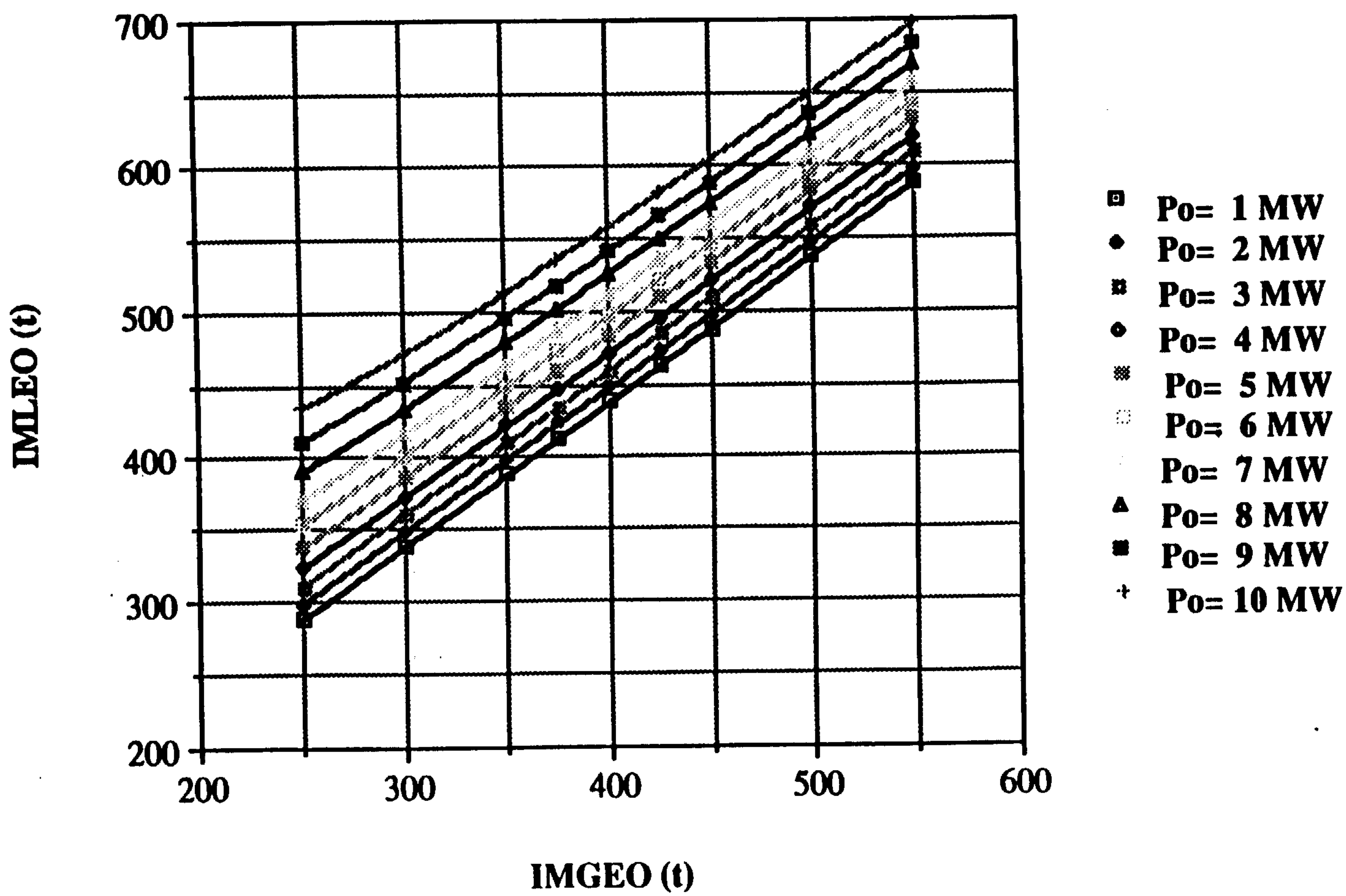


Solar Array Mass Trade for Earth Spiral

The following graph shows the equivalent mass in LEO for the transfer array scenario vs. what the vehicle weighs at GEO. This weight in GEO would be the same if a chemical stage had boosted the vehicle from GEO. The trade was performed for different power levels to determine the most advantageous solution. To determine the correct power level, one must also take into account the time associated with the power level.

Solar Array Mass Trade for Earth Spiral

LEO to GEO Electric Spiral Analysis



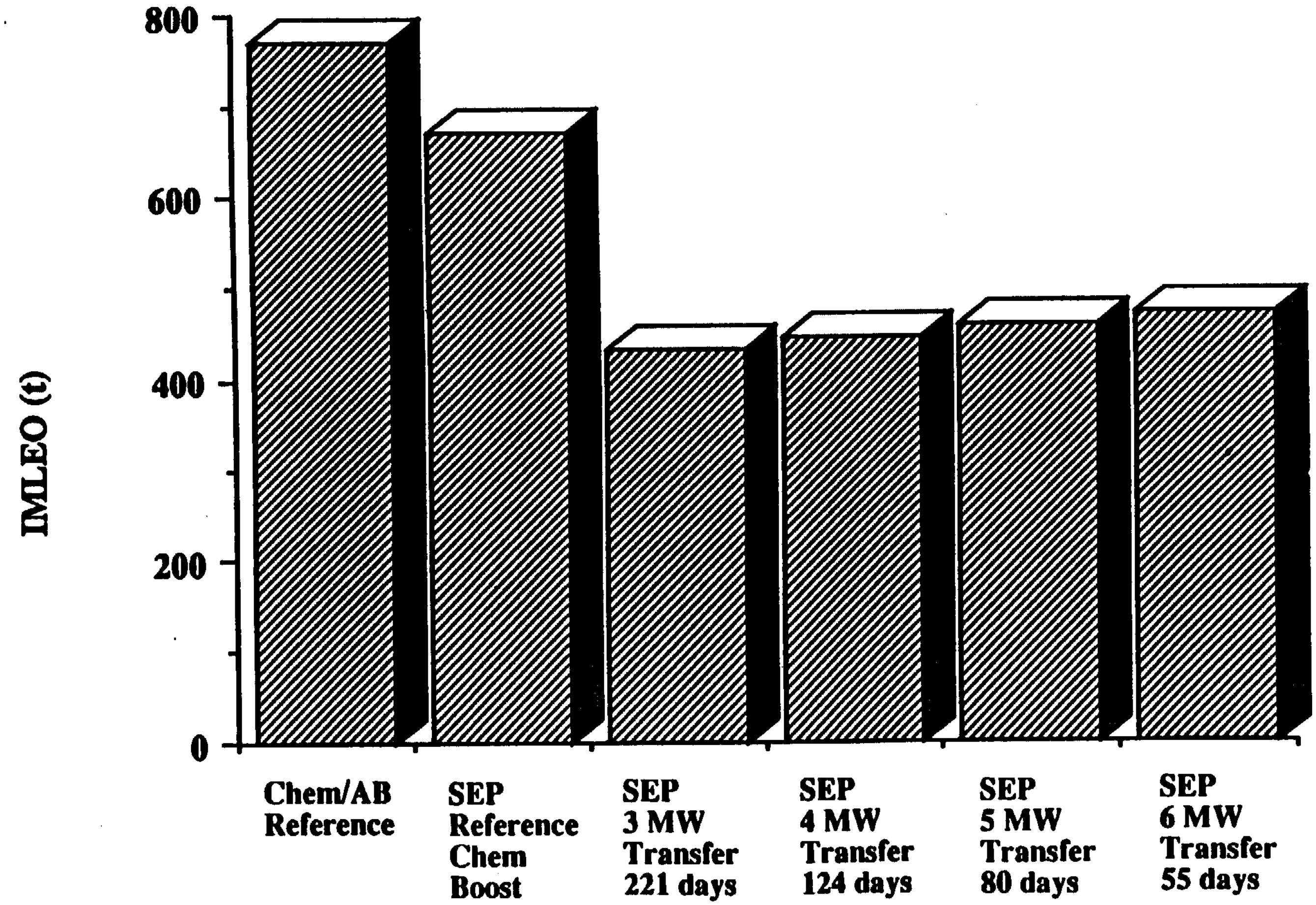
Mass of Expendable Array Scenarios vs. Reference Cases

The bar graph is a summary of the two preceding charts containing data on the spiral times and LEO mass for a given mass at GEO. From these charts it seems that a power level between 3 and 6 MWe would provide the best combination of weight savings and transfer times. For a reference point, 5 MWe has been chosen.

Mass of Expendable Array Scenarios Vs. Reference Cases

STCAEM/brc/9Feb90

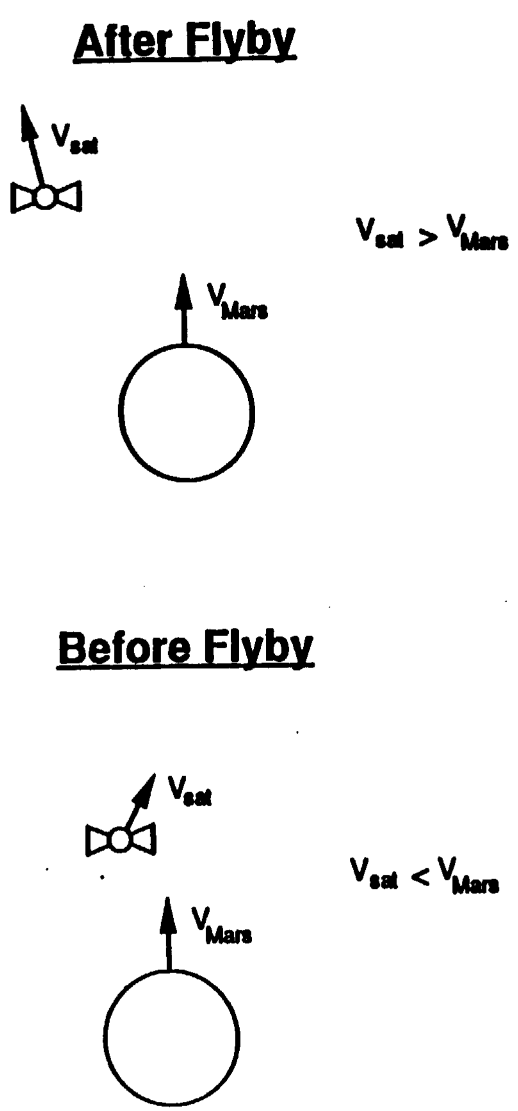
Assumes a 10 MW, 375 t (@ GEO) SEP Vehicle

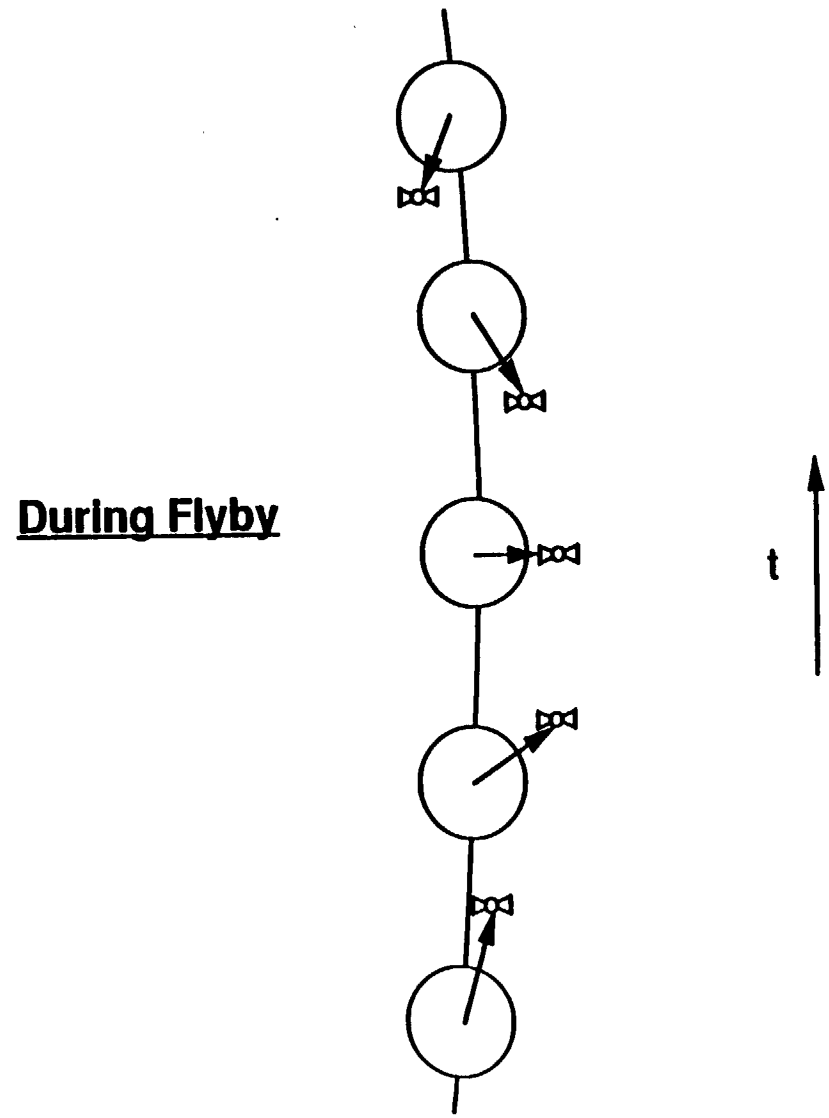


Mars Flyby: Inertial Reference Frame

The Mars flyby is shown for a SEP vehicle (Vsat) in the inertial reference frame. Before the flyby, the SEP vehicle is travelling slower than Mars. During the flyby, the vehicle flies in front of the planet, then past the planet allowing the planet to pass the vehicle. When the planet passes the vehicle (approximately 30 days), the vehicle flies past the planet, picking up a gravity boost, therefore reducing the trip time. During this scenario the vehicle does not spiral about the planet as in the reference case.

Mars Flyby: Inertial Reference Frame



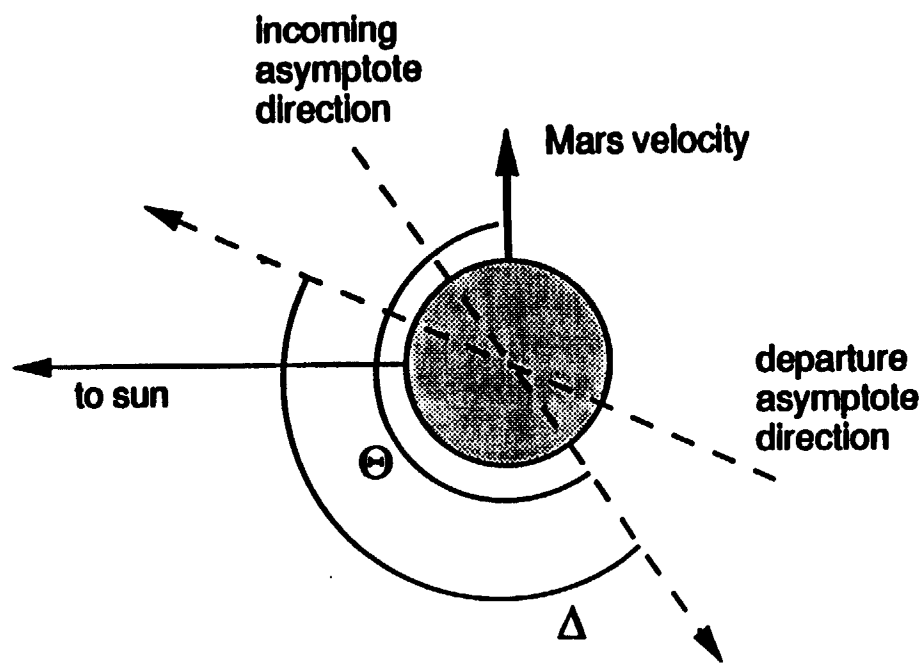


Mars Flyby: Mars Reference Frame

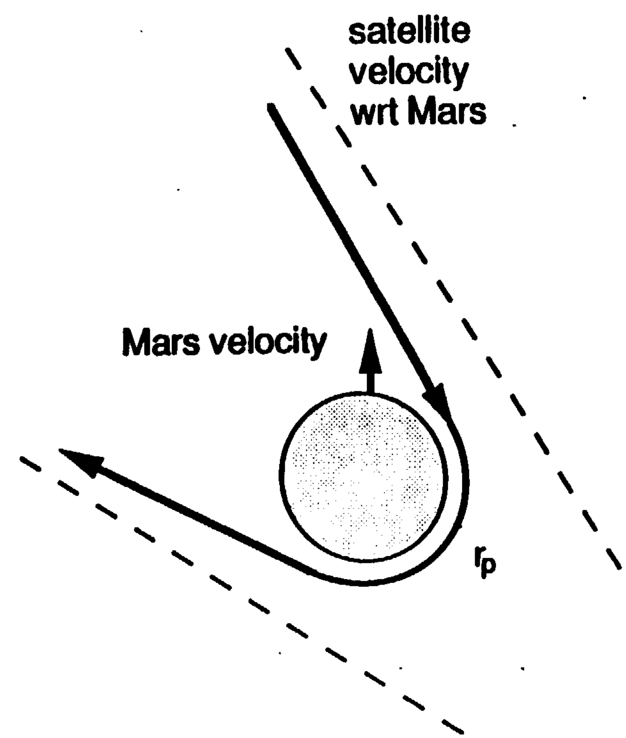
The following figures depict the Mars flyby from the Mars reference frame. The views show what the SEP vehicle would appear to be doing from the surface of Mars. During the 30 day stay time, the SEP vehicle would not enter an orbit about Mars, but would appear to be almost stationary.

Mars Flyby: Mars Reference Frame

Flyby Parameters



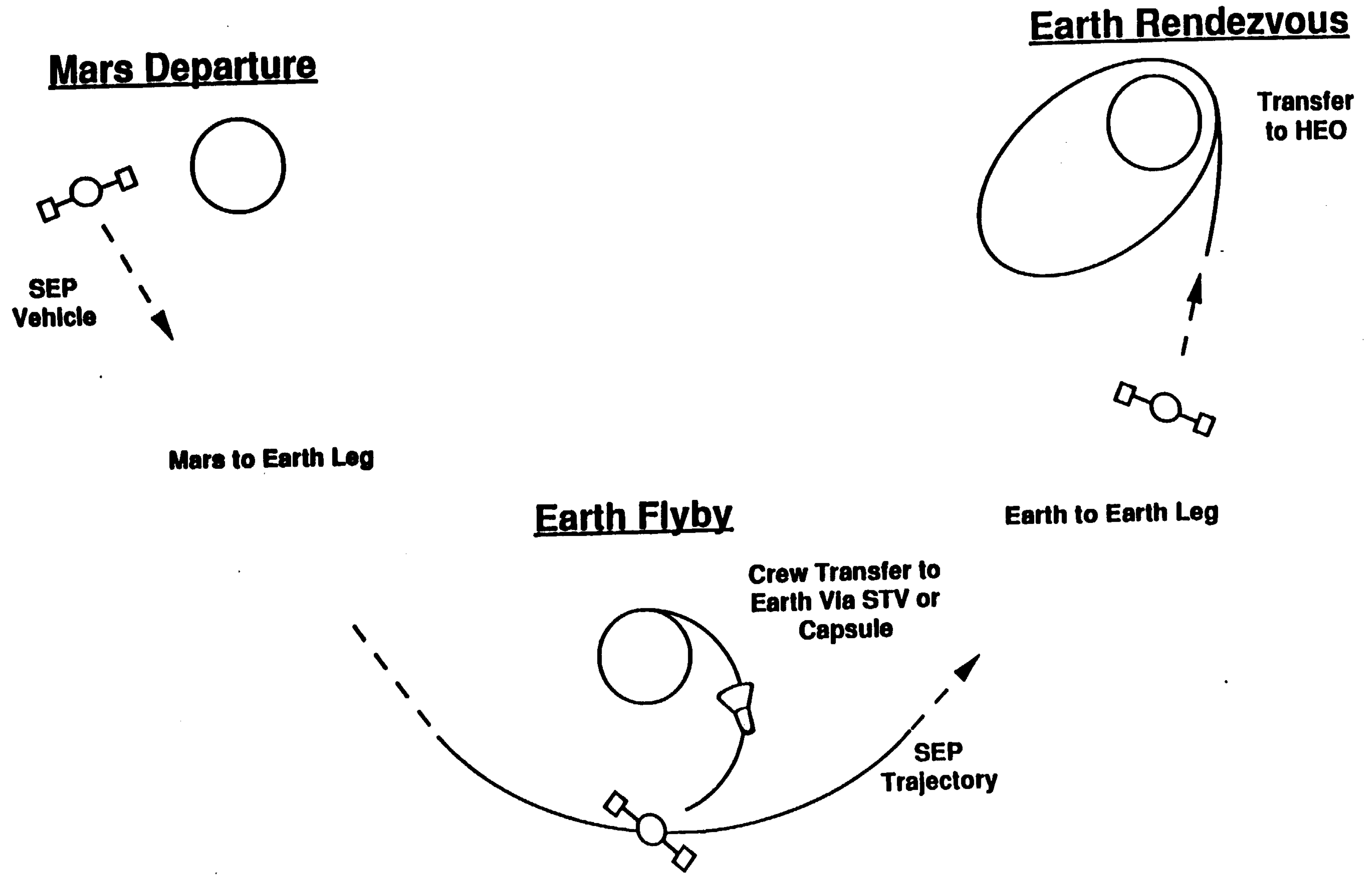
View from Mars Reference Frame



Earth Flyby

The advantage gained by an Earth flyby is due to the vehicle being able to accelerate for a longer period of time, before it has to decelerate. The vehicle will approach Earth with an excess speed limited to 5 km/sec. The vehicle will drop the crew off at earth via a STV or ECCV. The vehicle will spend up to 200 days "catching back up with" the Earth. The amount of return time can be traded against time saved during manned flight. One issue is thruster lifetime, which might limit the time the vehicle can spend trying to rendezvous with the earth. The Earth flyby will allow for a reusable SEP.

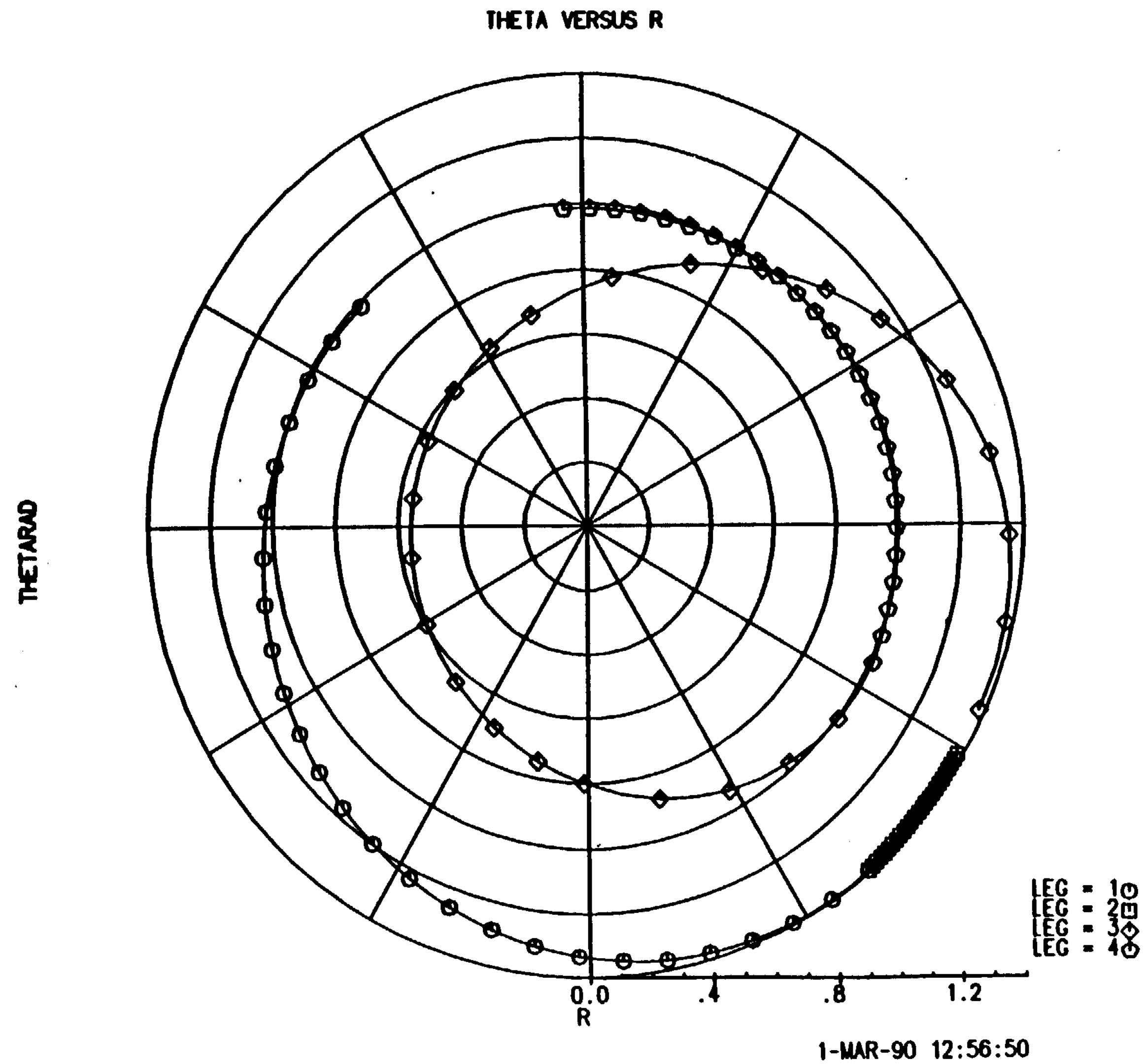
Earth Flyby


Mars Departure
SEP
Vehicle
Mars to Earth Leg
Earth Flyby
Crew Transfer to
Earth Via STV or
Capsule
SEP
Trajectory
Earth Rendezvous
Transfer
to HEO
Earth to Earth Leg

SEP Trajectory with Flybys

The trajectory plot combines the planetary flybys referenced previously and plots an actual trajectory generated by CHEBYTOP. The Mars stay and the Earth flyby and rendezvous are included.

SEP Trajectory with Flybys



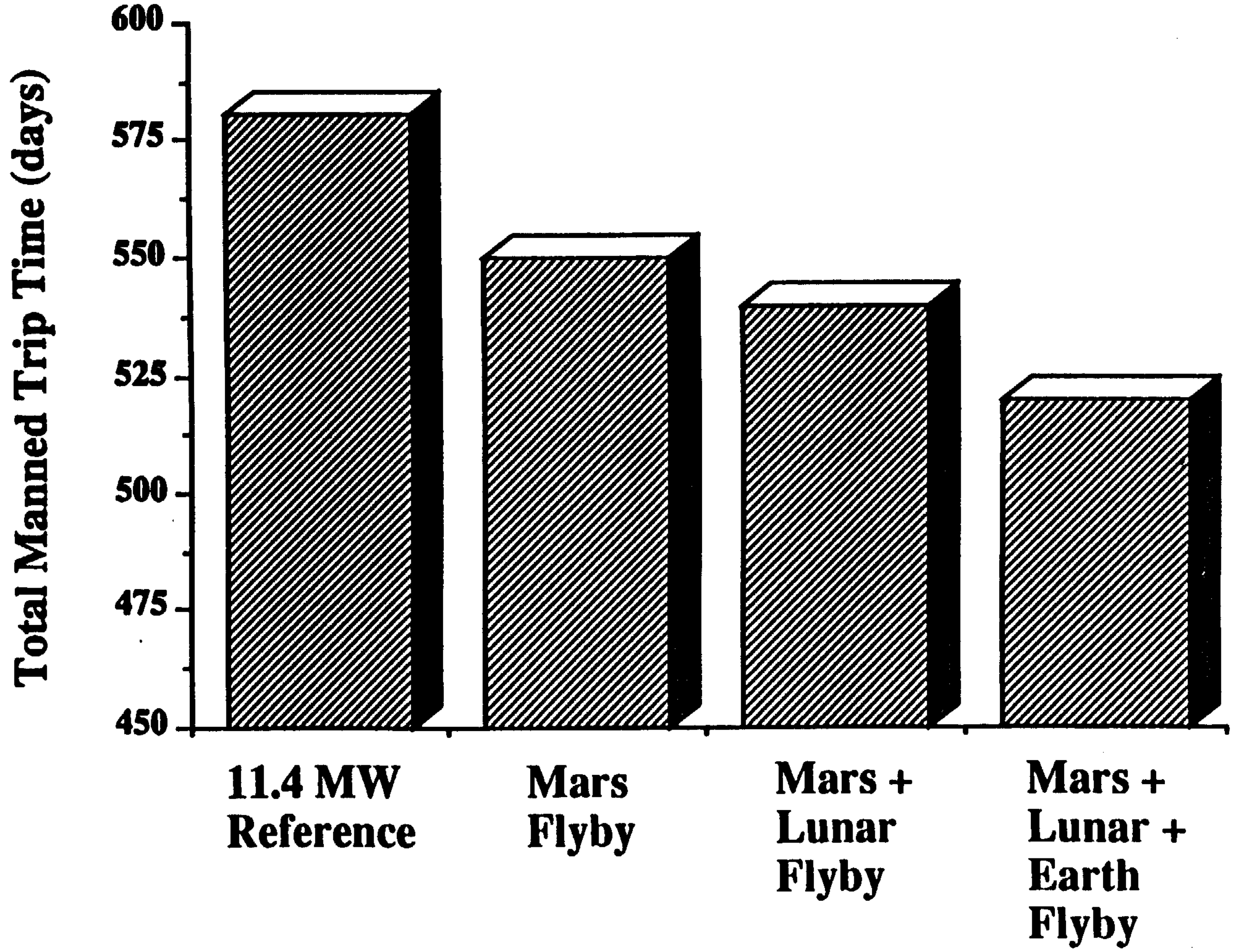
SEP Travel Time vs. Mission Type

The purpose of the swingby mission analysis was to decrease manned trip time for electric propulsion vehicles. Three different swingbys were analyzed that showed favorable results. A lunar, Earth, and Mars swingby showed preliminary benefits of trip time savings. A Venus swingby opportunity has not been found that would provide benefits for a low-thrust vehicle at this time.

The following graph shows a reference SEP vehicle and corresponding trip time for comparison purposes. The advantages or gains of the three different swingbys can be seen. A total manned trip time of 520 days (for the given vehicle) can be obtained, when all swingbys are employed.

SEP Travel Time vs. Mission Type

Time Comparison for SEP Missions Using Swingbys



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Orbital and Space-Based Requirements

Ernie Henshaw

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SPACE
SYSTEMS

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On-Orbit Assembly

- **Purpose**
  Define orbital and space-based support equipment, crew and facilities requirement/interfaces. By transportation element, for each scenario (Task 5-2).

- **Man Mars Vehicle Baseline**
  - Mars Excursion Vehicle
    - Aerobrake
    - Descent System
    - Ascent System
    - Mars Surface Payload
    - Mars Science Payload

  - Mars Transit Vehicle
    - Aerobrake
    - Trans Earth Injection System
    - Habitat Module

  - Trans Mars Injection System
    - Core Stack
    - Propellant Tank Set (3 Tanks Baseline)

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Groundrules/Assumptions

- Off-SSF assembly of MMV
- HLLV available for MMV launch
    - 4 flights per year
    - On-orbit stationkeeping (< 1 week)
- SSF-based OMV capable of maneuvering complete MMV subassemblies (i.e. MTV crew habitat)
- Maximize automation and robotics for assembly tasks
- MMV LEO departure date---Feb 2016
- MMV has high level of BIT/BITE

On-Orbit Assembly Baseline

The MMV assembly is completed at ET-derived MMV Assembly Platform (MAP). The MAP is constructed prior to MMV FEL. The MAP is self-supporting with power, control, and debris protection capability.

The MAP has line-of-sight communications with Space Station.

The MAP is out-fitted with a Space Station type Resource Node, which contains work stations for MAP and robotic local control, and a Space Station type Payload and Logistic Module for resupply of consumables and crew provisions.

The crew is transferred from Space Station by the OMV in the ACRV as required.

The MTV Habitat Module is first element launched to provide early crew quarters.

Crew is required for internal subsystem checkout, critical assembly monitoring and contingency operations.

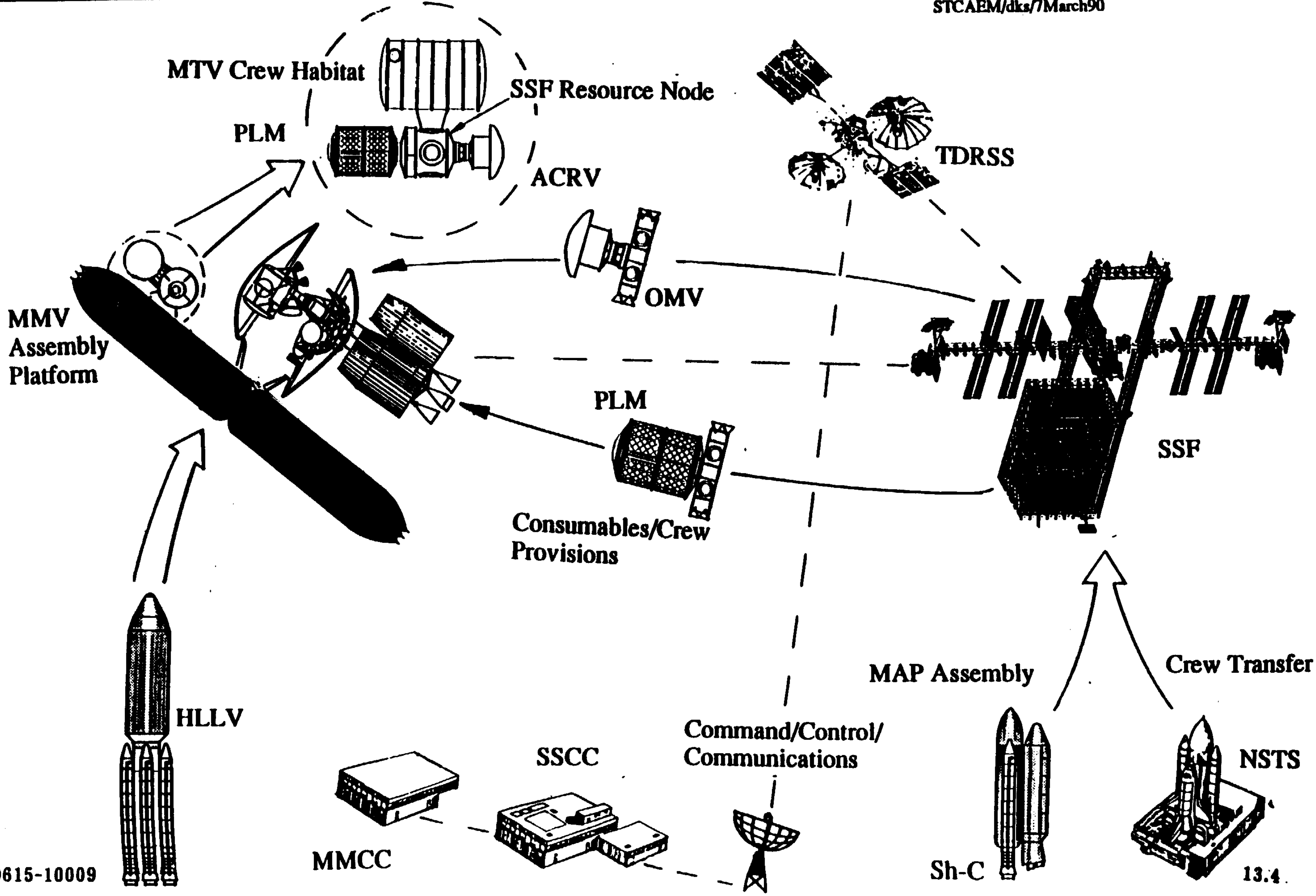
MAP - MMV Assembly Platform
OMV - Orbital Maneuvering Vehicle
ET - External Tank
FEL - First Element Launch
RMS - Remote Manipulator System
PRMS - Platform Remote Manipulator System
RAMS - Remote Aerobrake Manipulator System
PAS - Platform Anchor System
ASF - Assembly Storage Fixture
SSCC - Space Station Control Center
MMCC - Mars Mission Control Center
STS - Space Transportation System
TDRSS - Tracking and Data Relay Satellite System

ADVANCED CIVIL SPACE SYSTEMS

MMV On-Orbit Assembly Operations Baseline

BOEING

STCAEM/dks/7March90



Orbital Debris Environment

Space Station Freedom's baseline orbital debris environment generated protection requirements from data in 1985. In April 1989, new environment data showed an increase in the debris flux, 4-5 times greater than previous data. By the year 2016, the debris flux will be 5 times greater than that of 1989. Therefore, MMV debris environment will be 20 - 25 times worse than the environment which SSF generated it's protection requirements from.

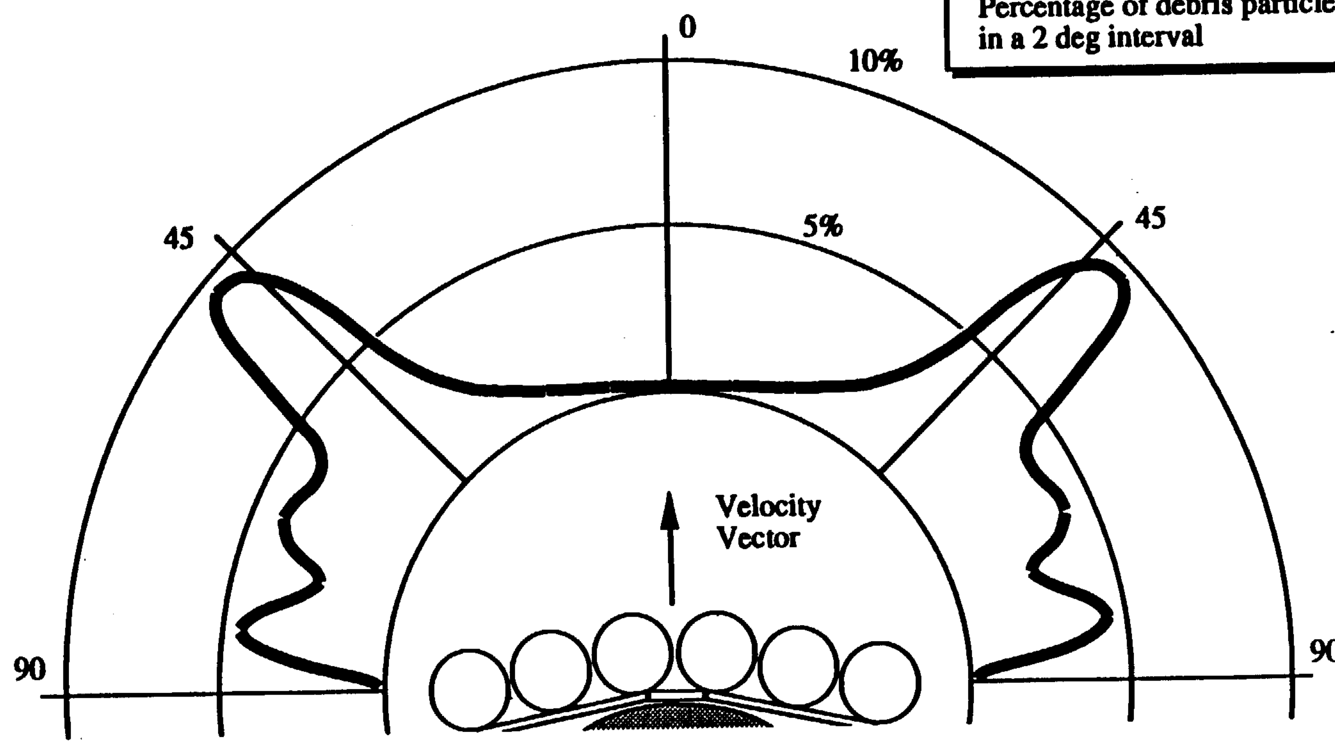
SSF has a current requirement of 99.55% probability of no penetration for each module for ten years.

SSF current shielding concept is an outer skin of 0.05 in. aluminum with a 4.3 in. void spacing between the outer skin and the 0.125 in. aluminum pressured wall.

The following page show graphically the debris flux relative to a velocity vector of a structure.

Orbital Debris Environment

Percentage of debris particles in a 2 deg interval



Normalized Closing Angle Density Function
500 km altitude, average 1990's environment

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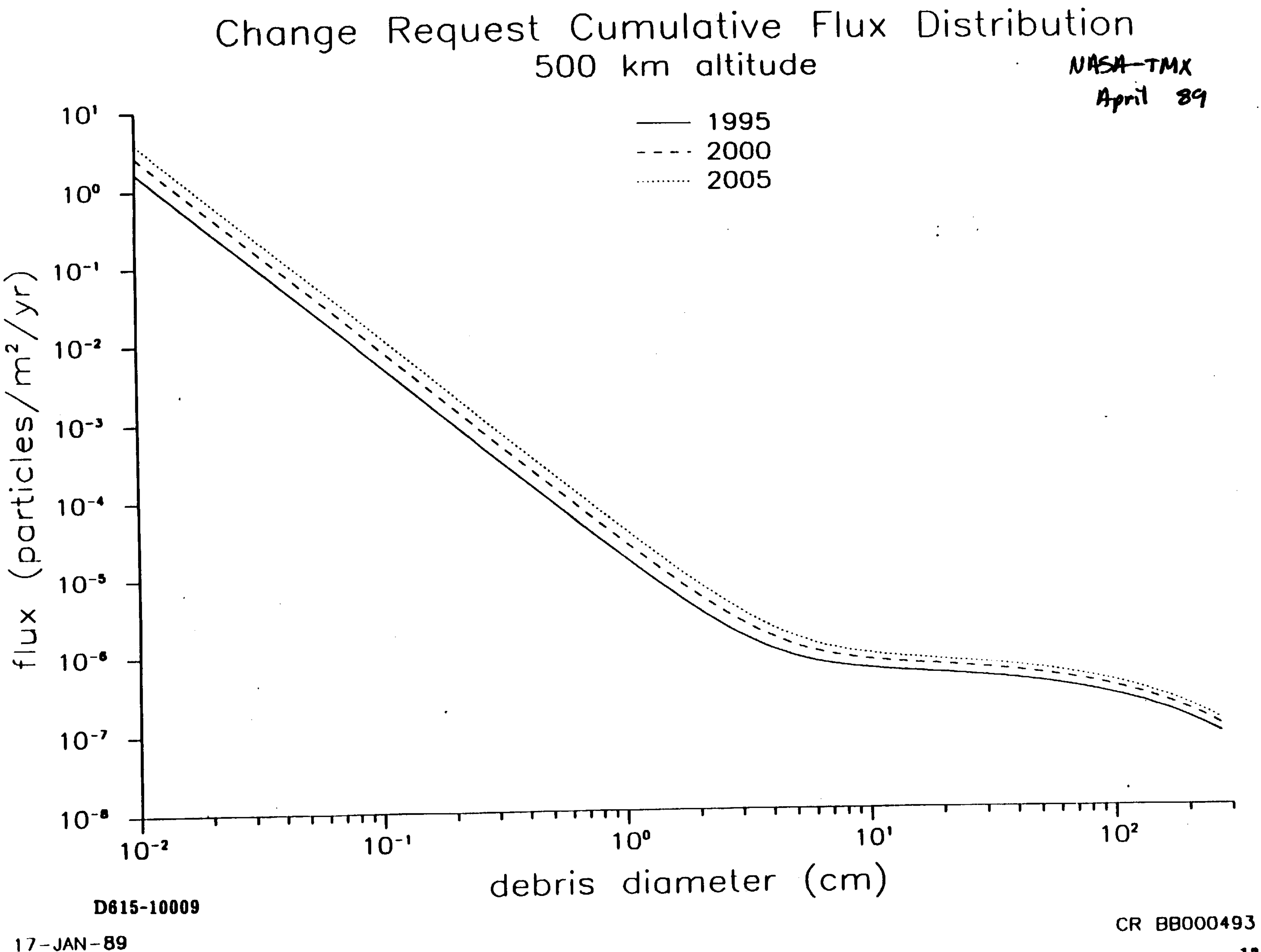

Change Request Cumulative Flux Distribution
500 km altitude
NASA-TMX
April 89
1995
2000
2005
flux (particles/m²/yr)
10¹
10⁰
10⁻¹
10⁻²
10⁻³
10⁻⁴
10⁻⁵
10⁻⁶
10⁻⁷
10⁻⁸
10⁻²
10⁻¹
10⁰
10¹
10²
debris diameter (cm)
D615-10009
17-JAN-89
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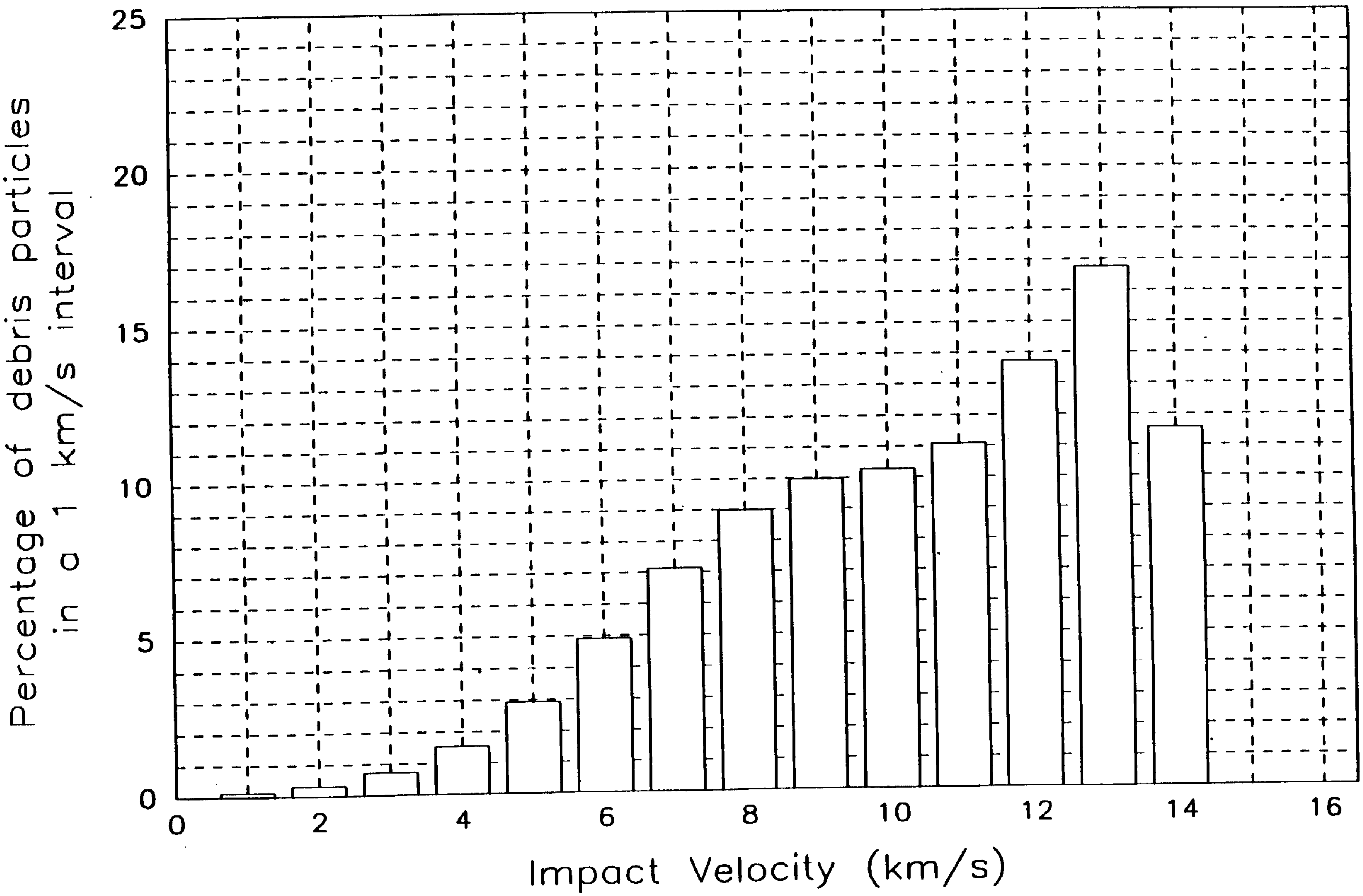
Normalized Closing Velocity Density Function

Change Request



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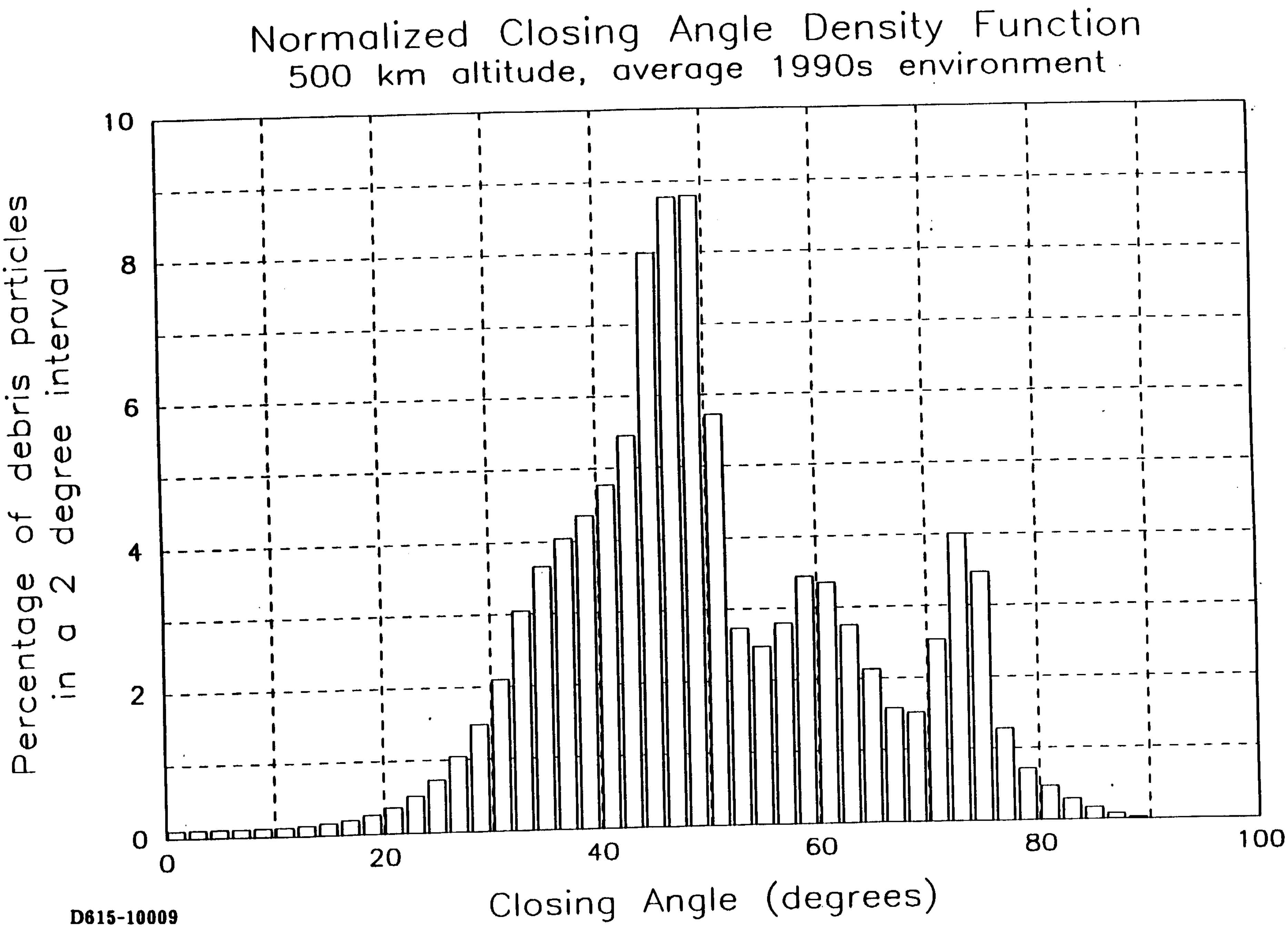

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22-DEC-89

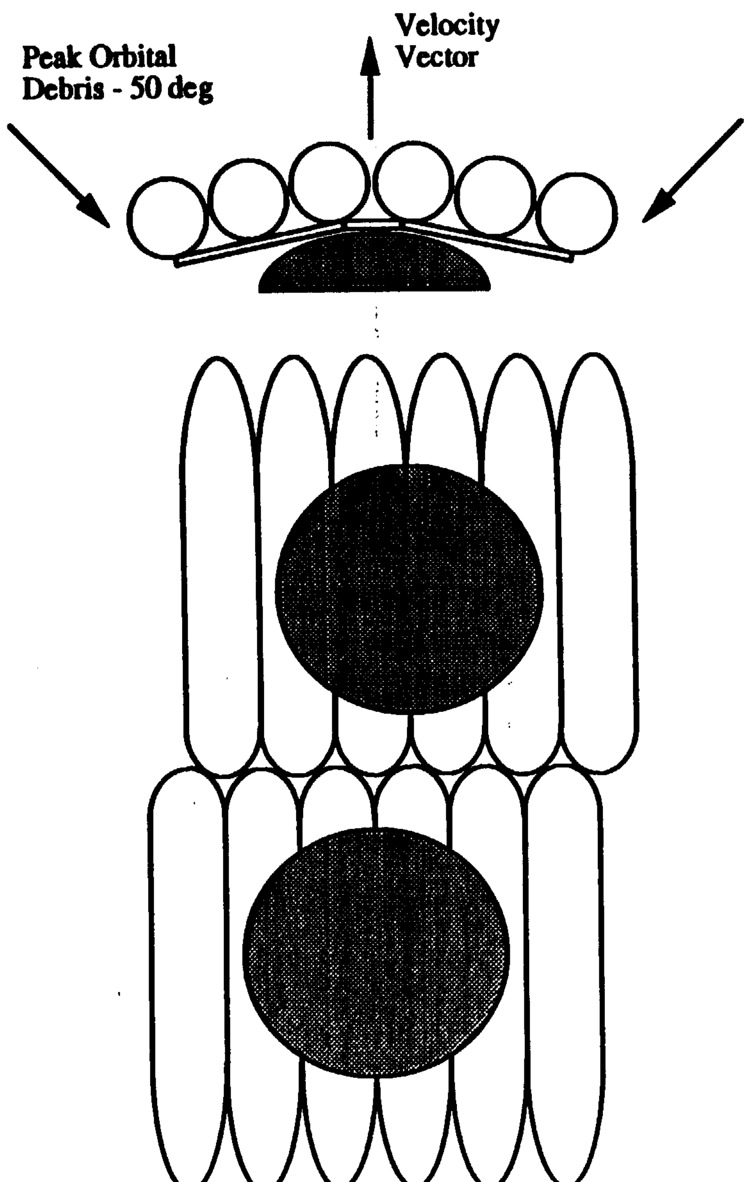
ET Debris Shield Concept for MMV Assembly

A concept was developed utilizing expended NSTS ETs on-orbitt to form a debris shield to provide protection and an assembly platform for the MMV. The concept requires two rows of six ETs attached by supporting structure. The platform will need power, guidance, attitude control, and boost subsystems. This concept shows the platform to be tilted into the velocity vector at 45 degrees to reduce drag.

This concept was developed to emphasis the requirement for debris protection. The concept allows the development of Assembly Support Equipment for any Debris Shield which provides protection during On-Orbit Assembly.

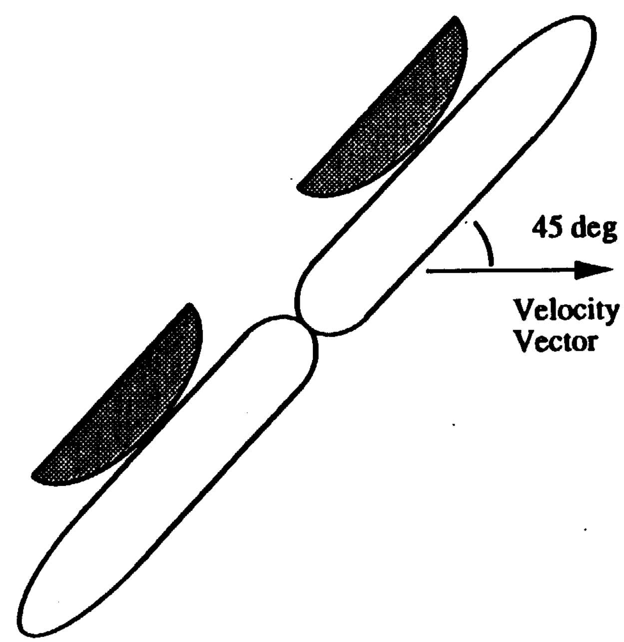
The following pages show the MMV Assembly Platform (MAP) with Aerobrake assemblies attached.

ET Debris Shield Concept for MMV Assembly



External Tank
- 47 x 8.4m dia
- "Raft" of 6 ET's provides a 47 x 50.4m shield
- "Raft" of 12 ET's provides a 94 x 50.4m shield

MTV/MEV Aerobrakes
- 30 x 27.4 x 7m thick



ET Debris Shield Concept

The following is a summary of advantages of the debris shield concept.

- One-third less weight penalty to orbit than separate shield -- 50K vs. 150K lbs.
- Provides greater protection than SSF shield design
- Debris shield can be completed years before MMV assembly start
- Provides experience with on-orbit assembly of large structures

The following is a summary of disadvantages of the debris shield concept.

- 3-4000 lbs penalty for each STS flight to orbit ET - 48K lbs total
  - 1000 lbs for vent, tumble valve, range safety system modifications
  - 2-3000 lbs for OMS propellant
- Additional 2000 lbs allowed for connecting structure
- High orbital drag
- May require on-orbit containment of SOFI
- Requires development of power, guidance, attitude control, reboost system

On-Orbit Assembly

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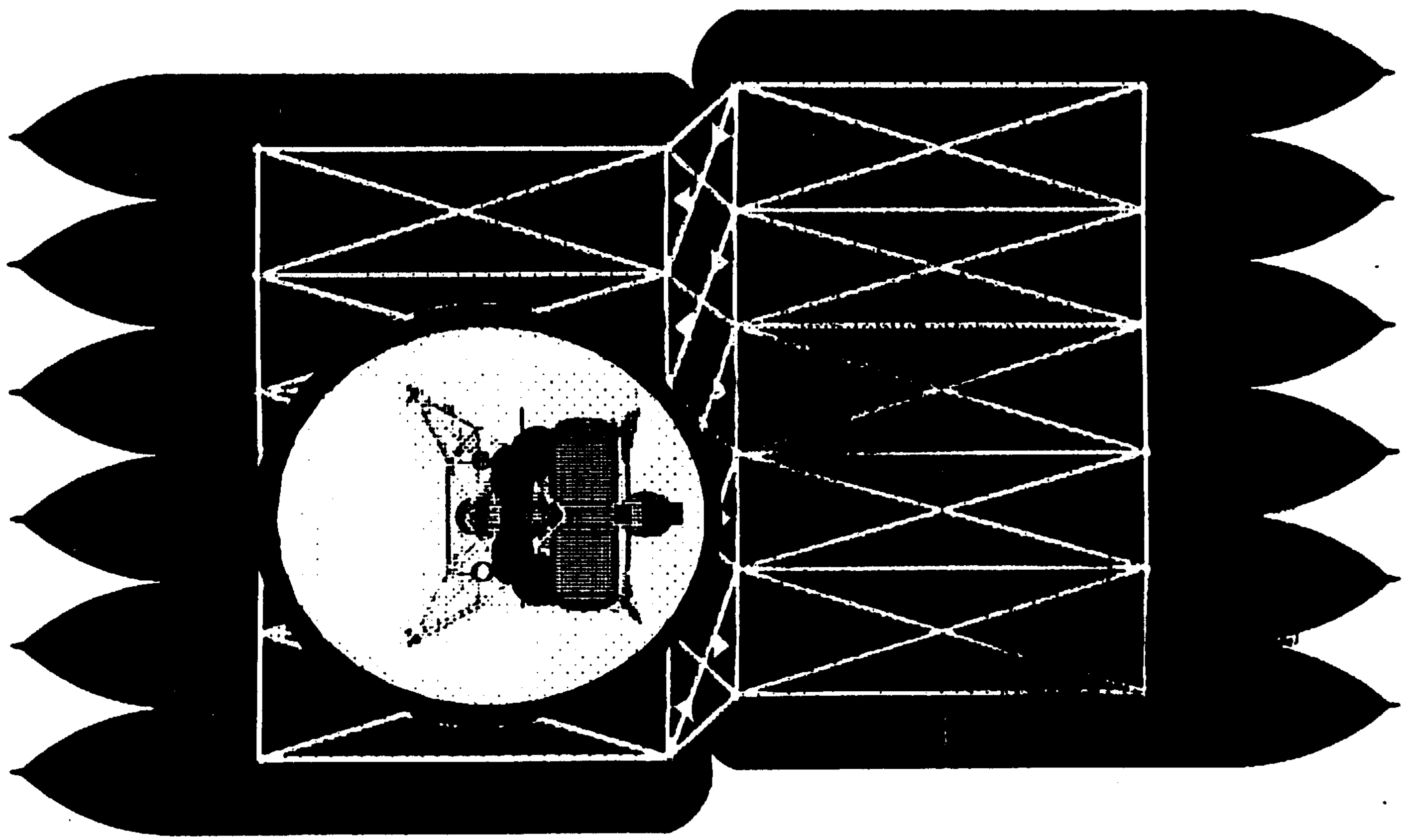
On-Orbit Assembly



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Off-SSF Assembly Node

Assembly Platform vs Integral MMV Assembly

	Advantages	Disadvantages
Assembly Platform	Available prior to MMV FEL Large debris protected area Platform for mounting power, control, communications subsystems Space for parallel assembly tasks/temporary storage of MMV components Platform for additional RMS	Separate vehicle to control (ground/ proximity operations) Additional launches required
Integral MMV Assembly	All required subsystems already available in some form Allows thorough checkout of subsystems prior to launch	Flight hardware used for debris protection Modification of flight power, control subsystems to control vehicle during assembly phase Requires storage space at SSF

Assembly platform provides large protected assembly workspace with minimum impact to MMV flight hardware

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ET Debris Shield Assembly Sequence

- First Orbiter/ET flight
  - First Orbiter/ET taken to MMV assembly orbit
  - Orbiter separates from ET, turns PLB toward ET
  - EVA or RMS attachment of EPS, GNC, RCS, C&T packages

- Subsequent Orbiter/ET flights
  - Orbiter/ET rendezvous with debris shield
  - Attach connecting structure to existing ET hardpoints
  - Orbiter separates from ET, attach ET to debris shield
  - Upgrade/relocate subsystems as debris shield buildup continues
  - Complete initial configuration of 6 ET's/final of 12 ET's

- First MMV assembly flight
  - HLLV/OMV rendezvous with ET debris shield/assembly platform
  - Upgrade subsystems as required for aging, damage
  - Install PRMS
  - Stow MMV components on assembly platform
  - Initiate MMV assembly

- Subsequent MMV assembly flights
  - HLLV/OMV rendezvous with ET debris shield/assembly platform
  - Stow MMV components
  - Continue MMV assembly
  - Upgrade/relocate subsystems as MMV buildup continues
  - Resupply consumables

MMV Manifesting

The following two pages show the manifesting analysis using the HLLV 10-meter diameter shroud as the transportation vehicle.

MMV Manifesting

- Vehicle: HLLV (2 or 3 Stage)
  - Abilities - 2 Stage
    - 10M x 30M Payload Envelope
    - 84 ton capacity
  - Abilities - 3 Stage
    - 7.6M x 30M Payload Envelope (less 3rd Stage)
    - 120 ton capacity
- HLLV Mission One (2 Stage)
  - MTV Habitat Module
  - Mars Surface Payload
  - Assembly Platform Support Equipment
- HLLV Mission Two (2 Stage)
  - MEV Aerobrake Sections
  - MTV Habitat Module Refurbishment/Consumables
- HLLV Mission Three (2 Stage)
  - MEV Aerobrake Sections
  - Assembly Platform Support Equipment

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MMV Manifesting (cont'd)

- HLLV Mission Four (2 Stage)
  - MEV Lander Structure
  - Lander Legs
  - Descent System
  - Ascent System
  - Science Payload
  - Airlock
  - Stairs

- HLLV Mission Five (2 Stage)
  - MTV Aerobrake Sections
  - MTV Habitat Module Consumables

- HLLV Mission Six (2 Stage)
  - MTV Aerobrake Sections
  - Assembly Platform Support Equipment

- HLLV Mission Seven (2 Stage)
  - MTV Trans Earth Injection System
  - MTV Habitat Consumables
  - Assembly Platform Support Equipment

- HLLV Mission Eight (3 Stage)
  - TMI Propellant with Engines

- HLLV Mission Nine thru Eleven (3 Stage)
  - TMI Propellant

MMV Manifesting

The following two pages show the manifesting analysis using the HLLV 12.5-meter diameter shroud as the transportation vehicle.

MMV Assembly
HLLV Manifest Assumptions

2 stage HLLV
- 12.5 x 30m payload envelope
- 140mt capacity

27.4 x 30 x 7m deep aerobrake
- Sliced into 3 pieces along longitudinal axis
- Stacked to a height of 9m

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MMV Assembly
HLLV Manifest

Flight	Manifest
1	MTV habitat module, MTV/MEV crew systems, ECCV, MTV/MEV structures, 25t surface payload, MEV ascent stage
2	MTV aerobrake, TEIS O2 tankage, assembly equipment
3	MEV aerobrake, MEV top-off fuel, top-off fuel support structure, assembly equipment
4	MEV descent stage, MTV habitat module consumables, assembly platform support equipment
5	TMI engines, propellant
6	TMI propellant
7	TMI propellant
8	TMI propellant

On - Orbit Assembly

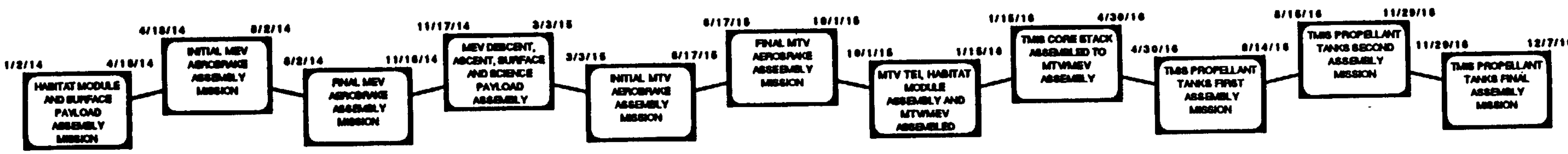
The preliminary Top Assembly Sequence is shown on the following page. The Man Mars Vehicle Assembly Platform (MAP) is operational prior to first MMV component launch to assembly orbit. Robotic assembly is primary mode of operation with crew participation only for contingency inspection and repair. Robotic operations will be monitored and controlled by the Ground Control Center. Assembly start date is February 28, 2013 to meet the departure from LEO date of February 2016.

The Assembly Sequences includes 90-day per mission processing time for the HLLV.

Note: The following Assembly Sequences are shown to three levels of depth (ie. Super Task, Task, and Subtask).

On-Orbit Assembly

MMV TOP ASSEMBLY



- Smallest unit of time is 1 hour
- 16 hours = 1 day of Assembly Duration

BASELINE DURATIONS:
- HLLV Launch = .5 day
- HLLV achieves stable orbit = .25 day
- OMV deploys from/to Freedom = .5 day
- OMV berths to components = .25 day
- Unstow and power up Robotics = .06 day
- Robotic verification = .12 day
- HLLV deploys components = .06 day
- OMV transfers components = .25 day
- Robotic tasks = .06 day
- EVA/Robotic Contingency = .5 day
- Component Inspection = .12 day
- Component Test = .25 day
- Subassemblies to stand-by mode = .5 day
- Mechanical Fastening of components = .18 day

On - Orbit Assembly

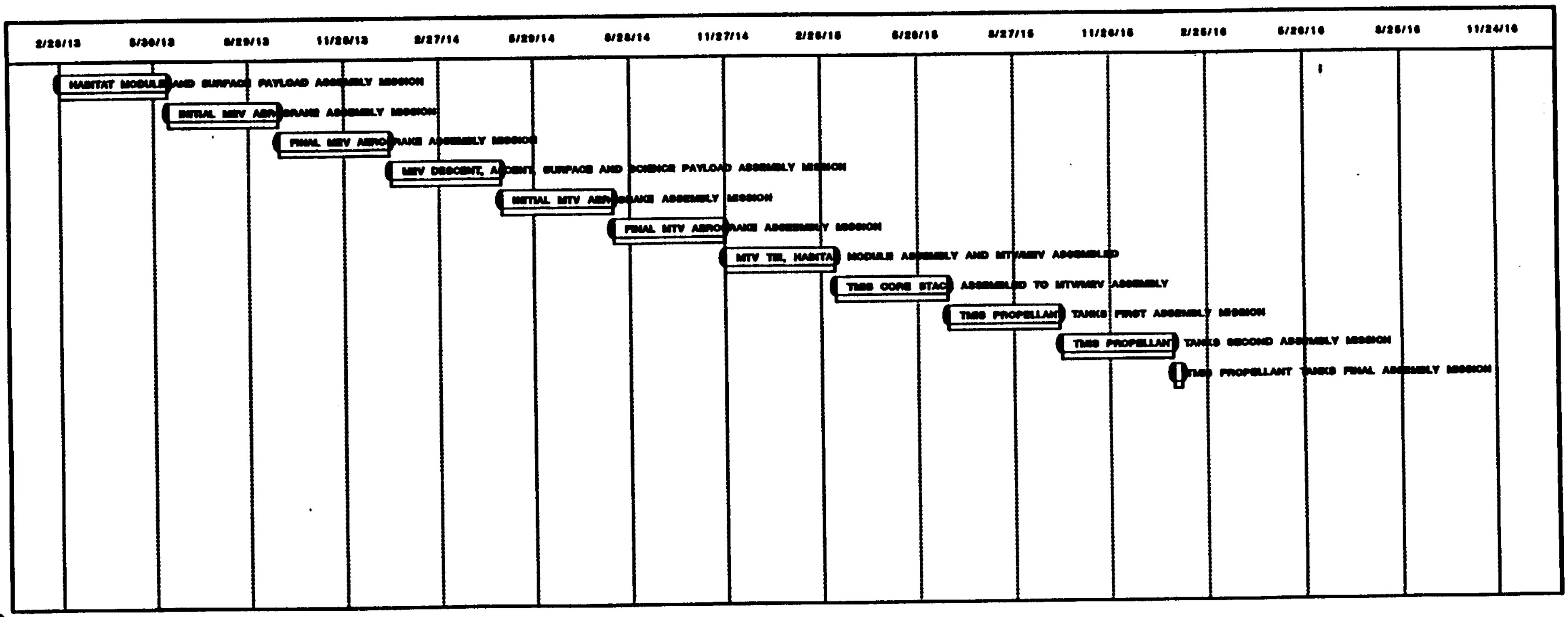
The following page shows the "Water-Fall" time schedule of the Top Assembly Sequence.

On-Orbit Assembly

ADVANCED CIVIL SPACE SYSTEMS

BOEING

STCAEM/dka/23Feb90



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On - Orbit Assembly

The following page shows the Project Schedule with Earliest Start, Earliest Finish, and Duration as assigned.

Note: The duration assigned are dependent on the 90-day HLLV processing time.

On-Orbit Assembly

Name	Earliest Start	Earliest Finish	Subproject	Days
HABITAT MODULE AND SURFACE PAYLOAD ASSEMBLY MISSION	2/28/13	6/14/13	HLLV MISSION ONE	106
INITIAL MEV AEROBRAKE ASSEMBLY MISSION	6/14/13	9/28/13	HLLV MISSION TWO	106
FINAL MEV AEROBRAKE ASSEMBLY MISSION	9/28/13	1/12/14	HLLV MISSION THREE	106
MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY	1/13/14	4/29/14	HLLV MISSION FOUR	106
INITIAL MTV AEROBRAKE ASSEMBLY MISSION	4/29/14	8/13/14	HLLV MISSION FIVE	106
FINAL MTV AEROBRAKE ASSEEMBLY MISSION	8/13/14	11/27/14	HLLV MISSION SIX	106
MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED	11/27/14	3/13/15	HLLV MISSION SEVEN	106
TMIS CORE STACK ASSEMBLED TO MTV/MEV ASSEMBLY	3/13/15	6/27/15	HLLV MISSION EIGHT	106
TMIS PROPELLANT TANKS FIRST ASSEMBLY MISSION	6/27/15	10/11/15	HLLV MISSION NINE	106
TMIS PROPELLANT TANKS SECOND ASSEMBLY MISSION	10/12/15	1/26/16	HLLV MISSION TEN	106
TMIS PROPELLANT TANKS FINAL ASSEMBLY MISSION	1/26/16	2/3/16	HLLV MISSION ELEVEN	8

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On - Orbit Assembly

For this analysis, the HLLV, as described previously, was the only launch vehicle to provide transportation for the MMV components to the MAP. As shown on the following page, assembly time of the MMV itself is approximately four (4) man-months.

On-Orbit Assembly

ADVANCED CIVIL SPACE SYSTEMS

BOEING

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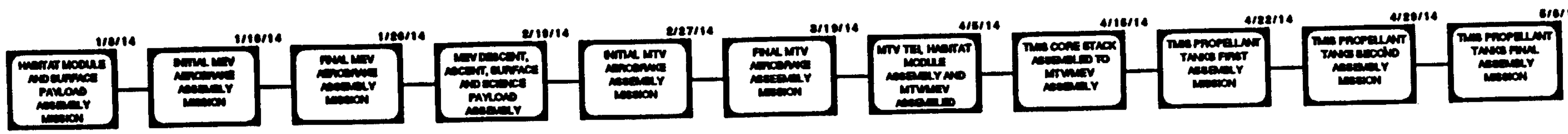
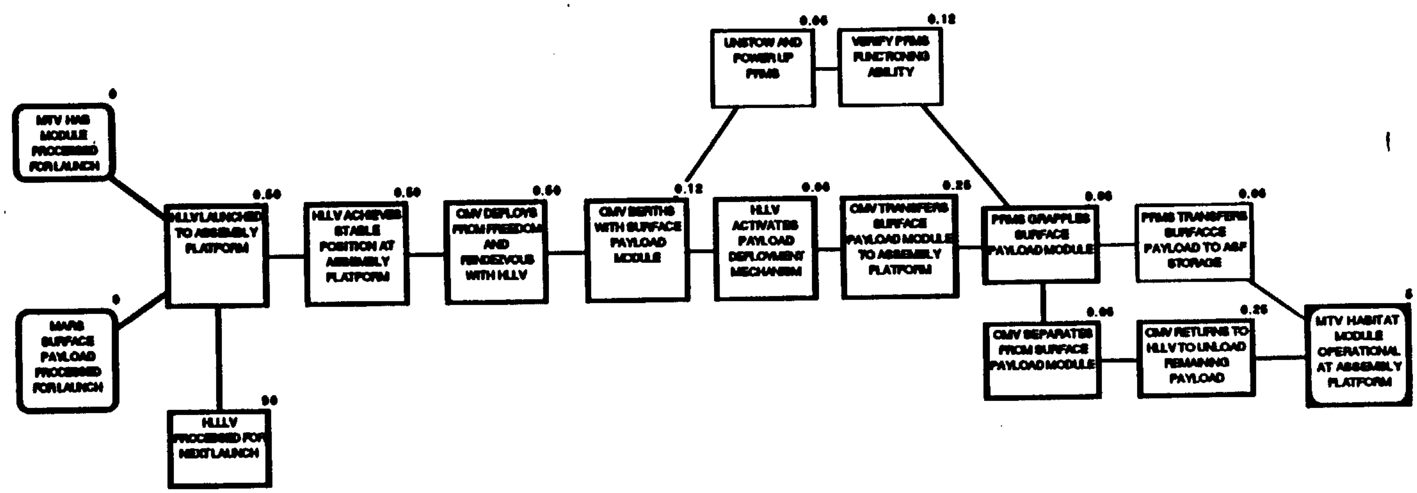


On - Orbit Assembly

The first assembly mission is shown on the following two pages. The MTV Habitat Module and the Mars Surface Payload Module are launched to the MAP orbit. The OMV transfers the components from the HLLV to the MAP. Mars Surface Payload Module is stored at an ASF for future assembly operations. The Habitat Module is berthed to the Assembly Node and readied for crew support.

HLLV MISSION ONE



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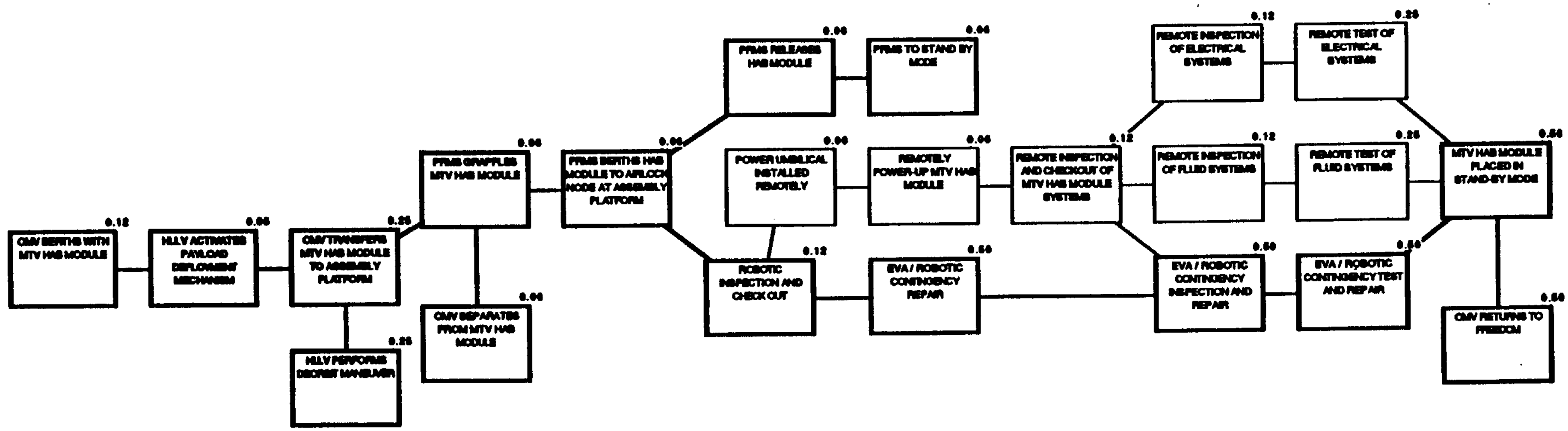
MTV HABITAT MODULE



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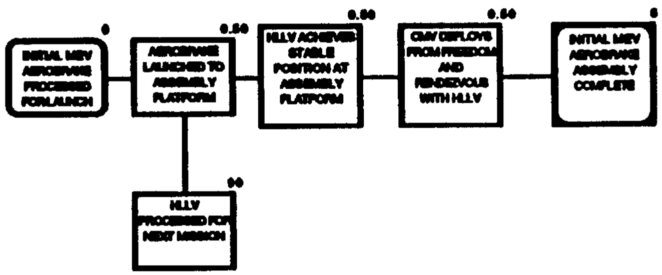
On - Orbit Assembly

The MEV Aerobrake is launched in two (2) HLLV missions, five (5) sections of Aerobrake per mission. The assembly sequence for the MEV Aerobrake is shown on the following nine pages.

The RAMS are launched in the first assembly mission to complete the precision mechanical fastening of the joints as required.

The TPS for the joints is installed during the final Aerobrake assembly mission.

HLLV MISSION TWO



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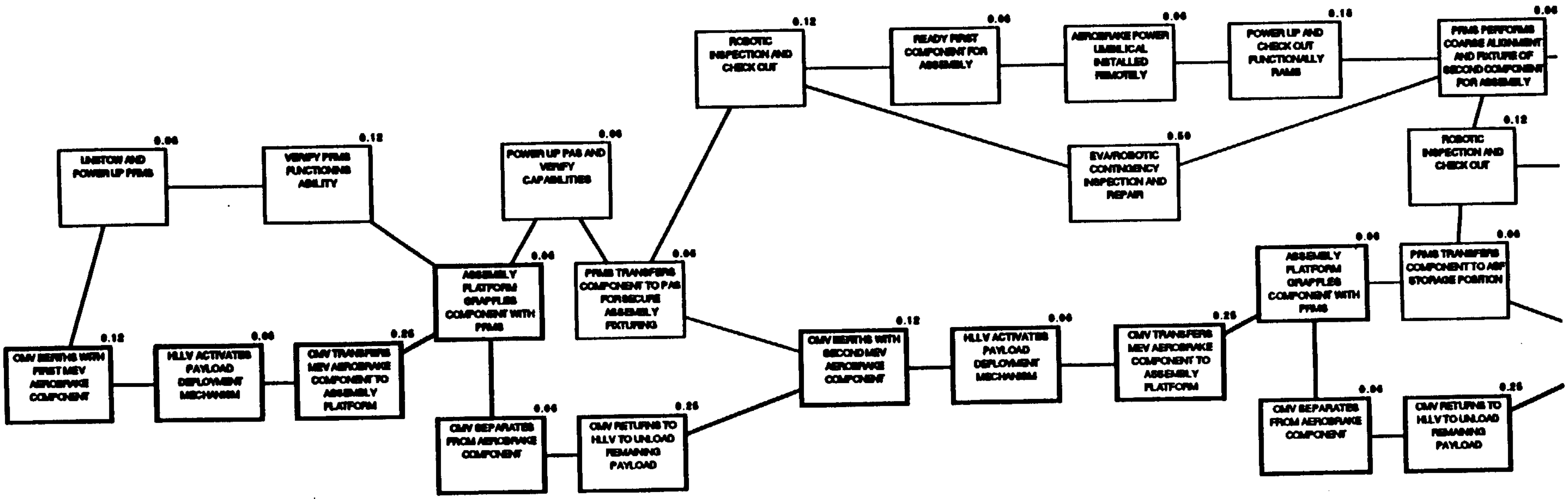
MEV AEROBRAKE MISSION ONE



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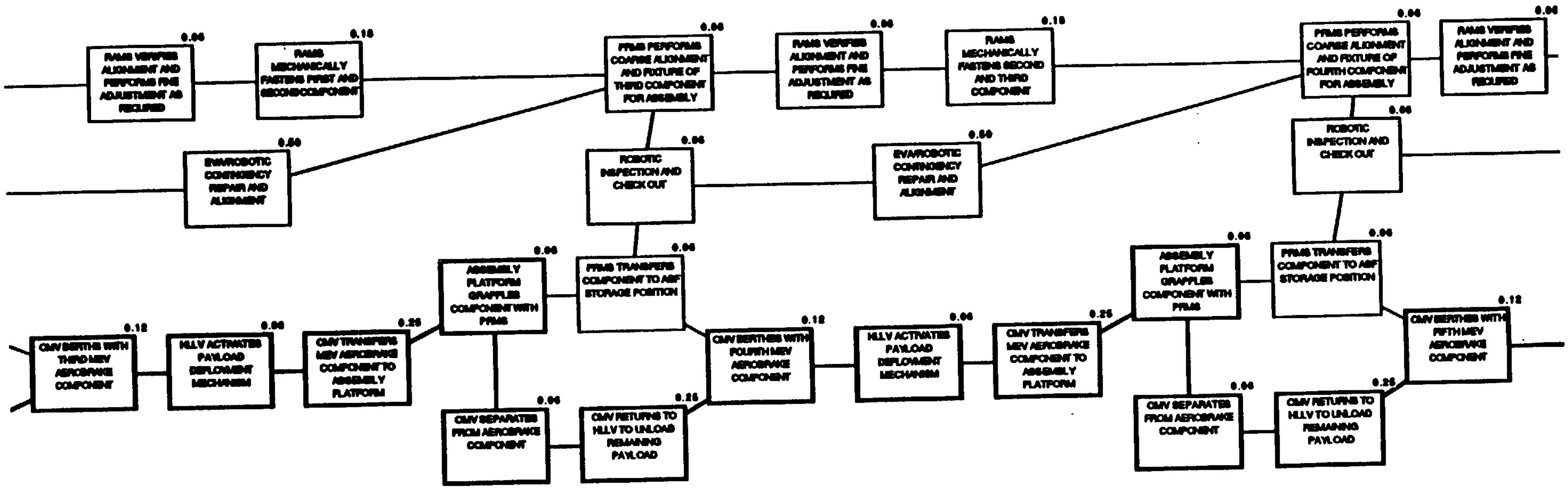

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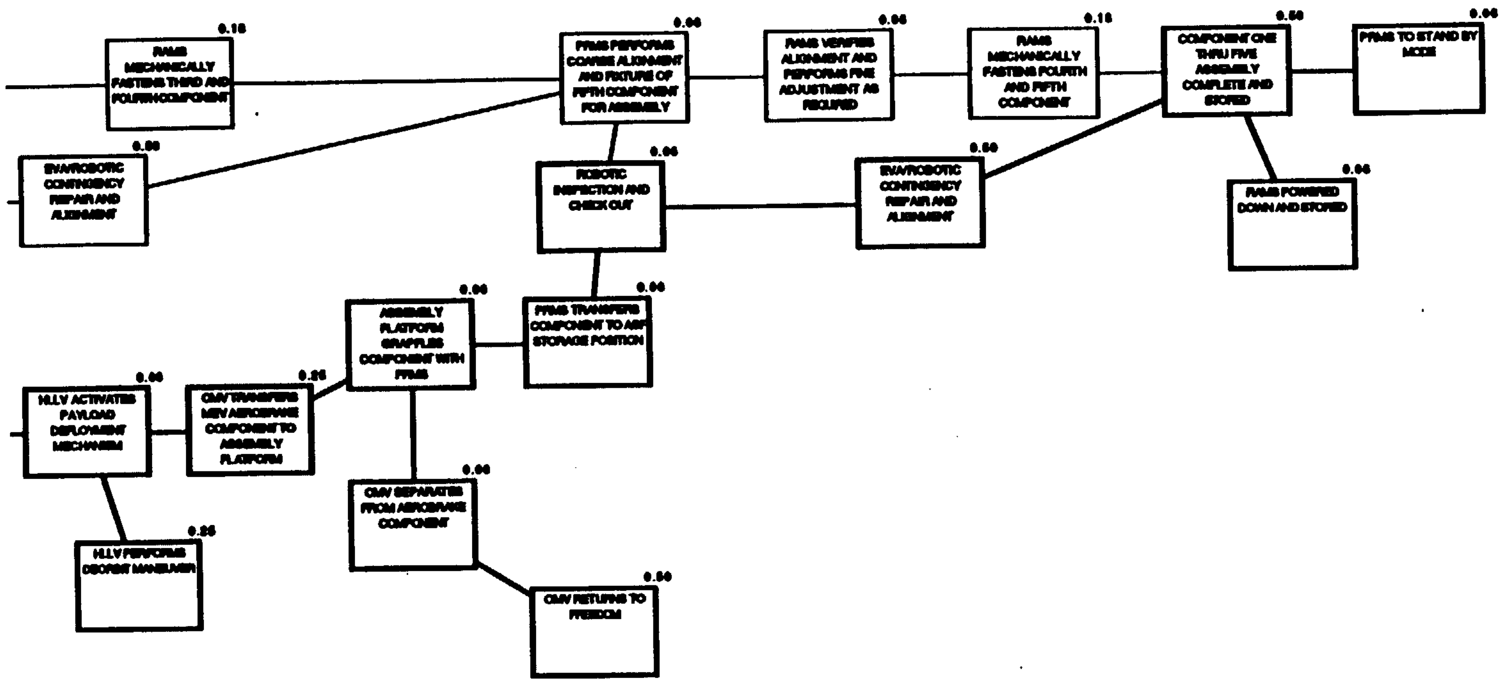

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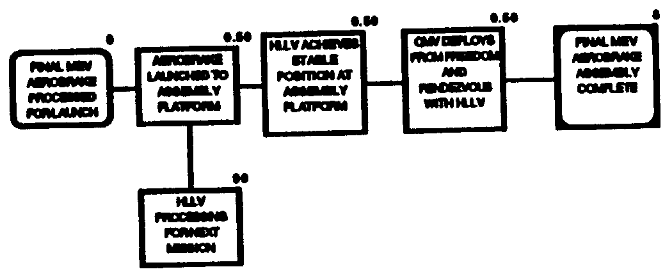
HLLV MISSION THREE



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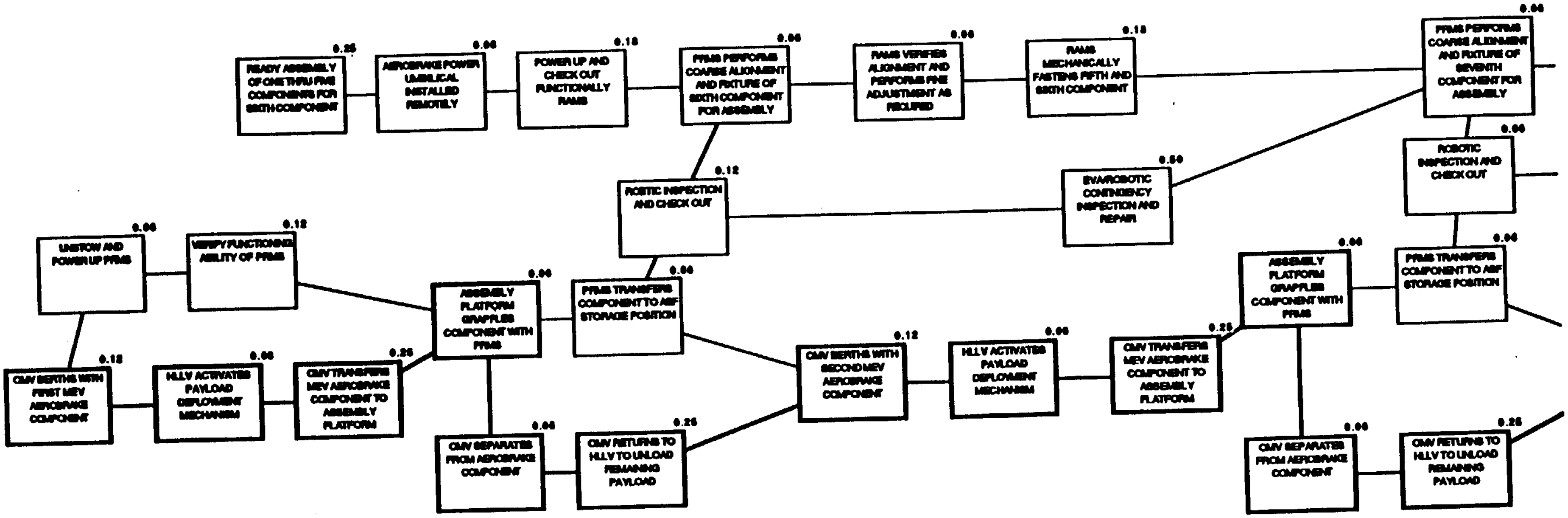
MEV AEROBRAKE MISSION TWO



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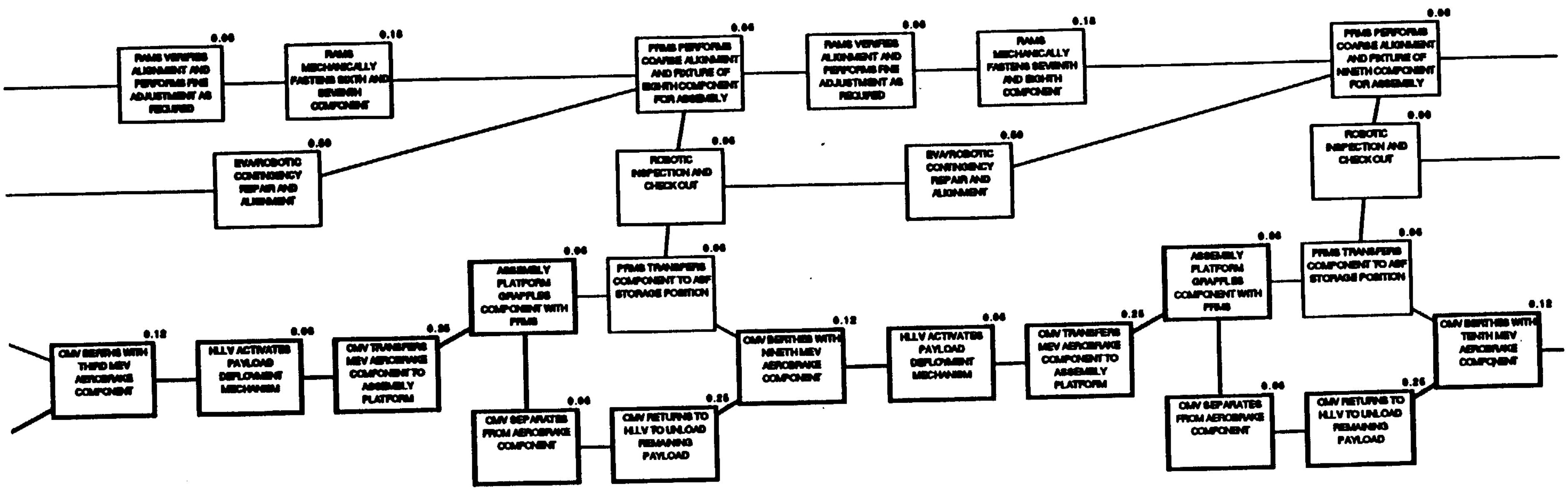
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
RAMS MECHANICALLY FASTENS SIXTH AND SEVENTH COMPONENT
0.18
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
FRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF EIGHTH COMPONENT FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.06
FRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
RAMS MECHANICALLY FASTENS SEVENTH AND EIGHTH COMPONENT
0.18
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
FRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF NINETH COMPONENT FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.06
FRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
OMV BERTHS WITH THIRD MEV AEROBRAKE COMPONENT
0.12
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH FRMS
0.06
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
OMV BERTHES WITH NINETH MEV AEROBRAKE COMPONENT
0.12
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH FRMS
0.06
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
OMV BERTHES WITH TENTH MEV AEROBRAKE COMPONENT
0.12

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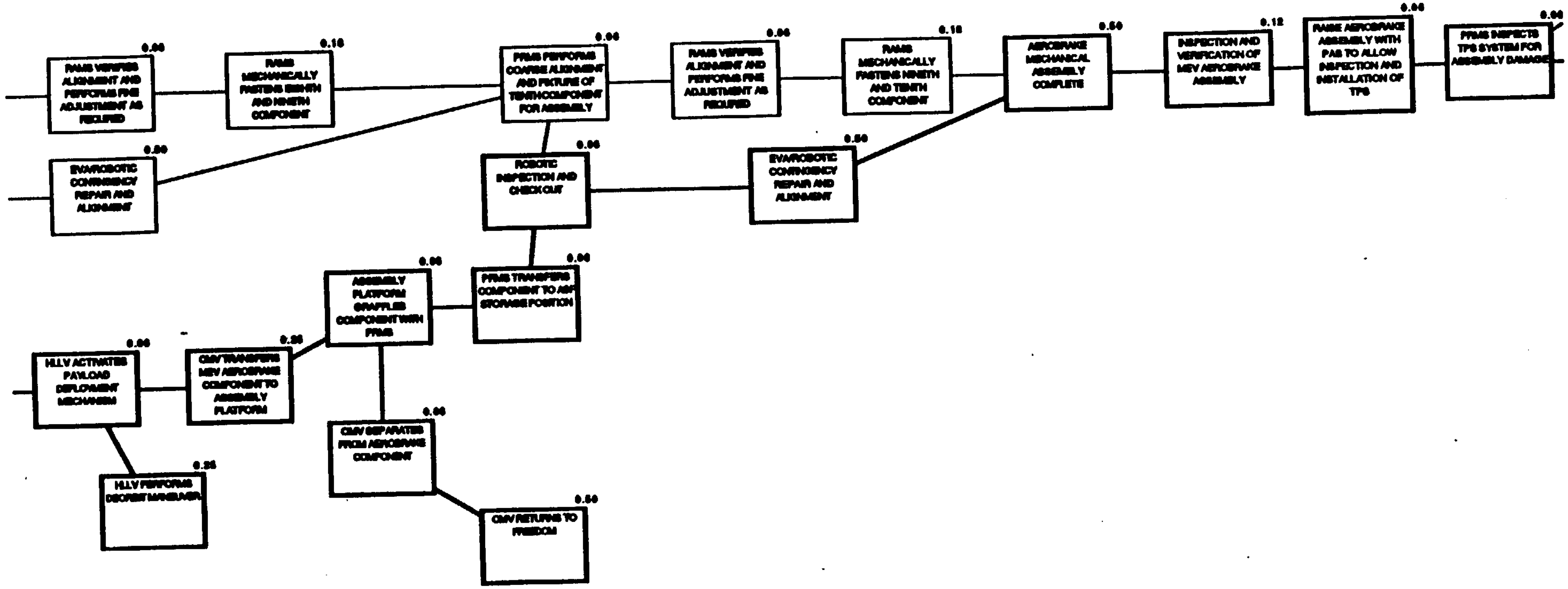

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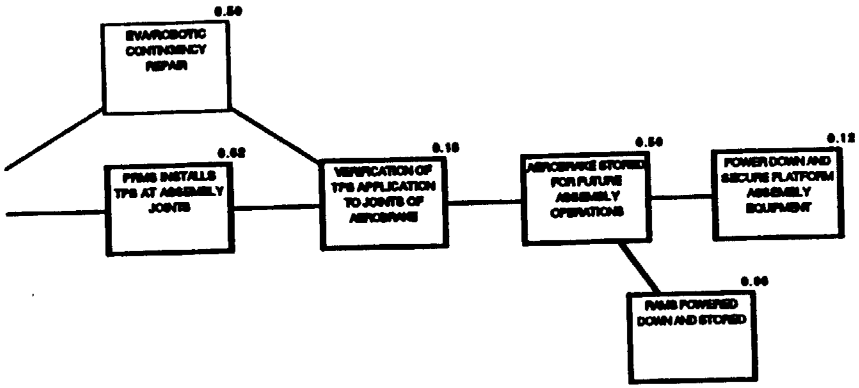

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On - Orbit Assembly

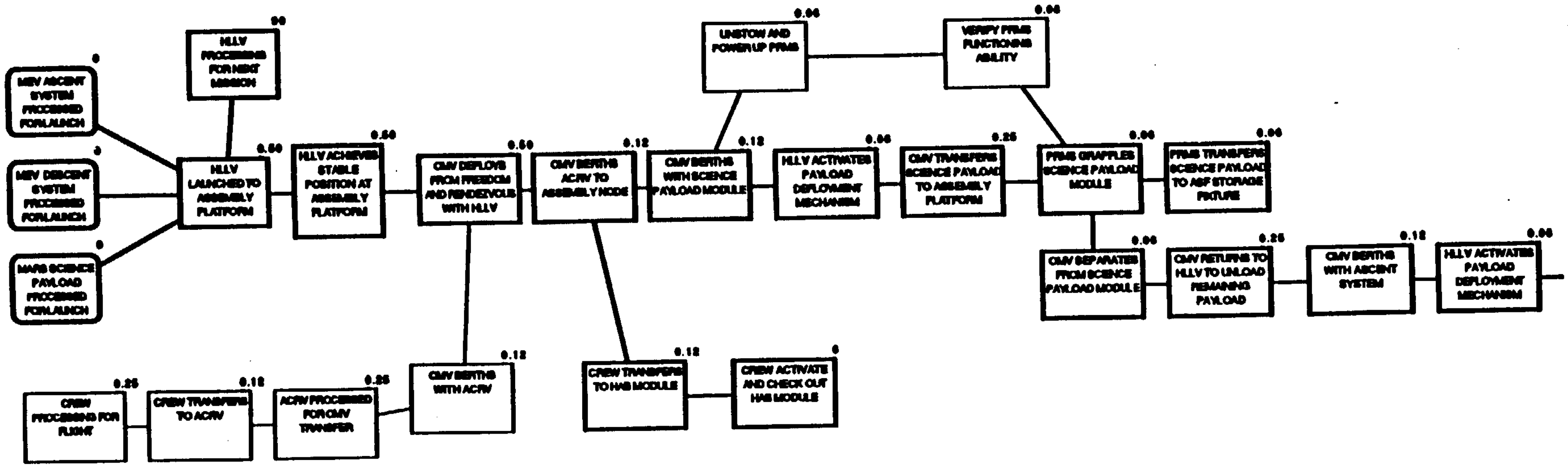
The MEV Assembly Sequence is shown in the following two pages. The Descent Lander System, Ascent System, and Science Payloads are launched to the assembly orbit.

The OMV transfers the assembly crew from Space Station to MAP in the ACRV. The ACRV is berthed to the Airlock/Node Assembly and the crew transfers to the Habitat Module.

The OMV transfers the Science Payload and the Ascent System to an ASF positioned on the MAP.

The Descent Lander System is transferred to the MAP by the OMV.

HLLV MISSION FOUR



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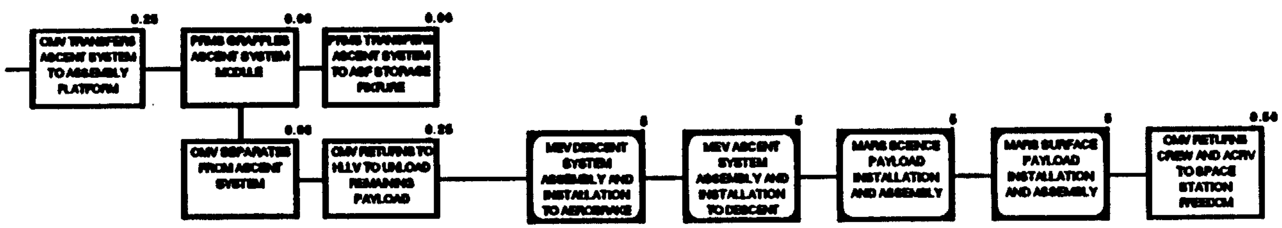
On-Orbit Assembly



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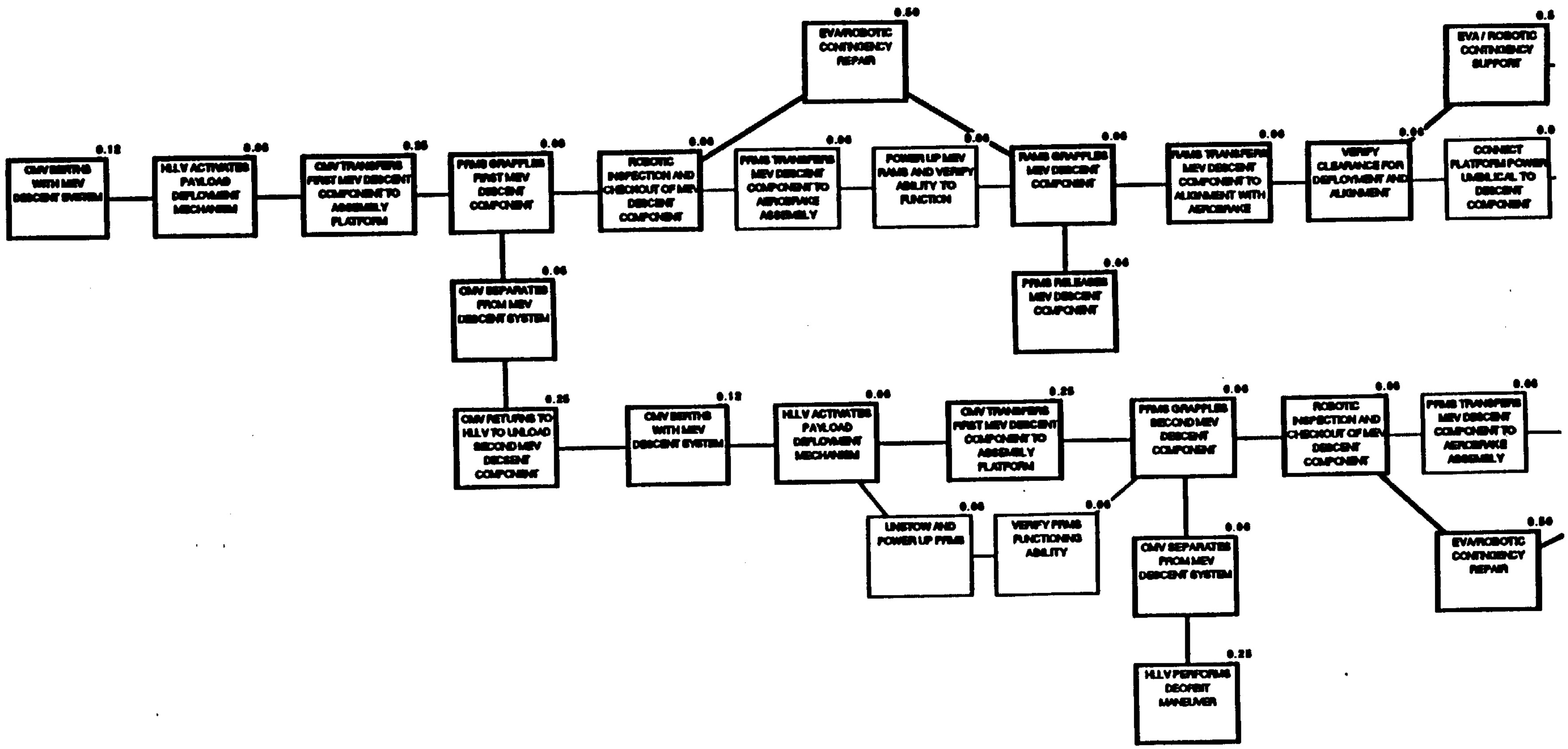
On - Orbit Assembly

The assembly and installation of the Descent Lander System is shown on the following three pages. The Descent Lander System is assembled in place to the MEV Aerobrake. Assembly crew monitors operations from the Habitat Module and performs contingency operations as required.

On-Orbit Assembly

MEV DESCENT SYSTEM



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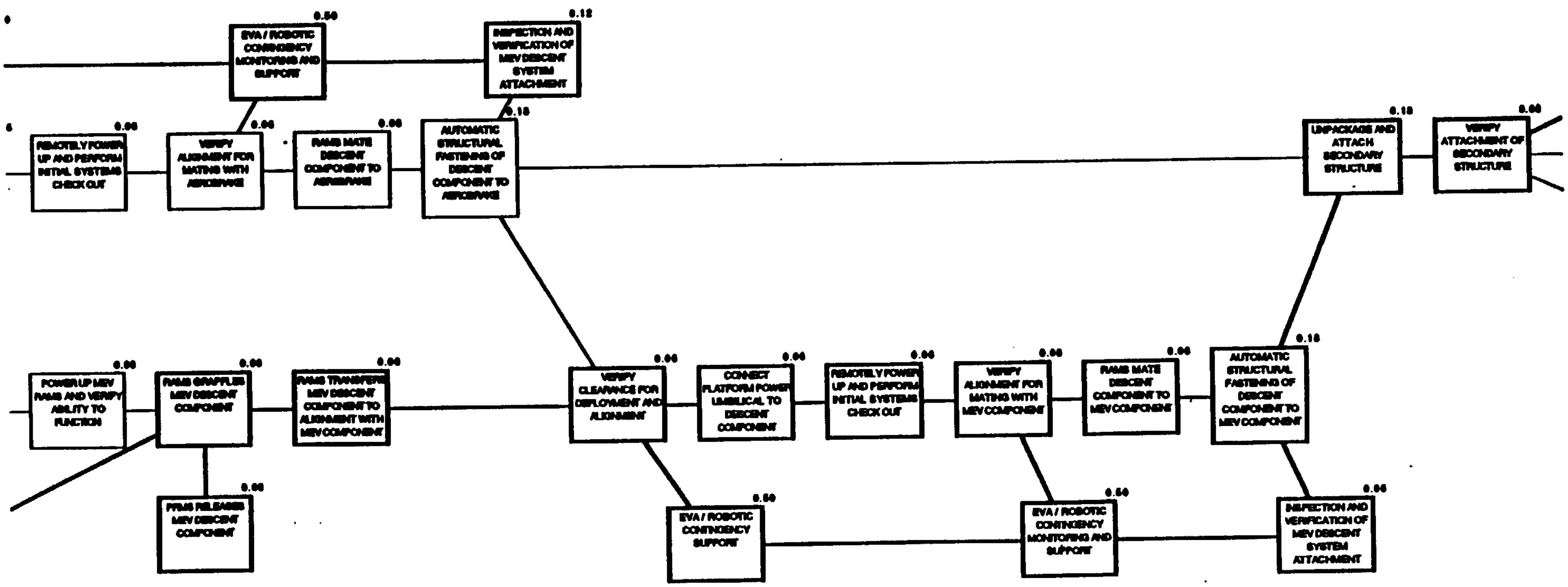
On-Orbit Assembly



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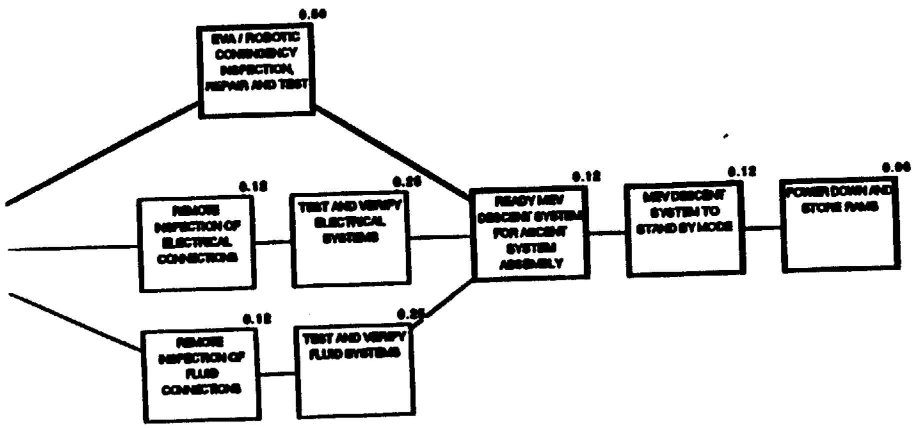
On-Orbit Assembly



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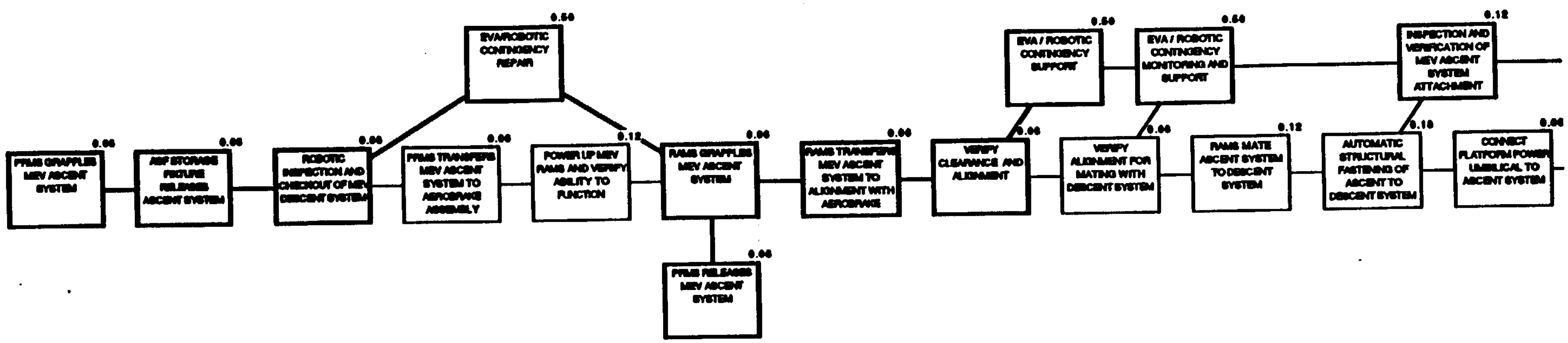
On - Orbit Assembly

The Ascent System installation to the Descent Lander System is shown in the following two pages. Assembly crew monitors operations and perform contingency task(s) as required.

MEV ASCENT SYSTEM



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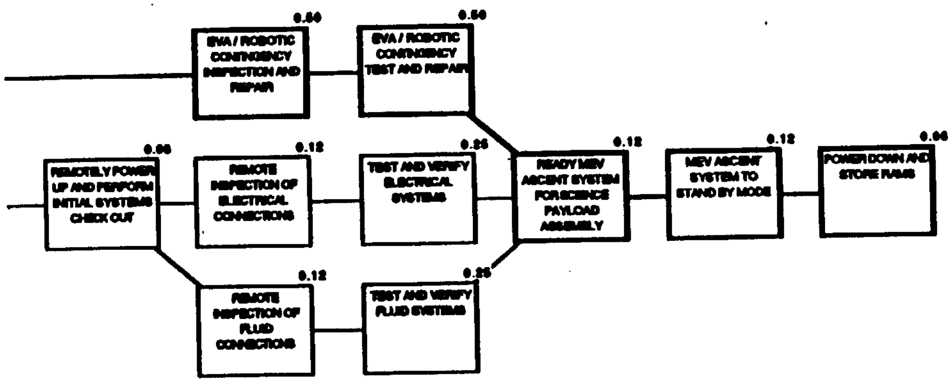
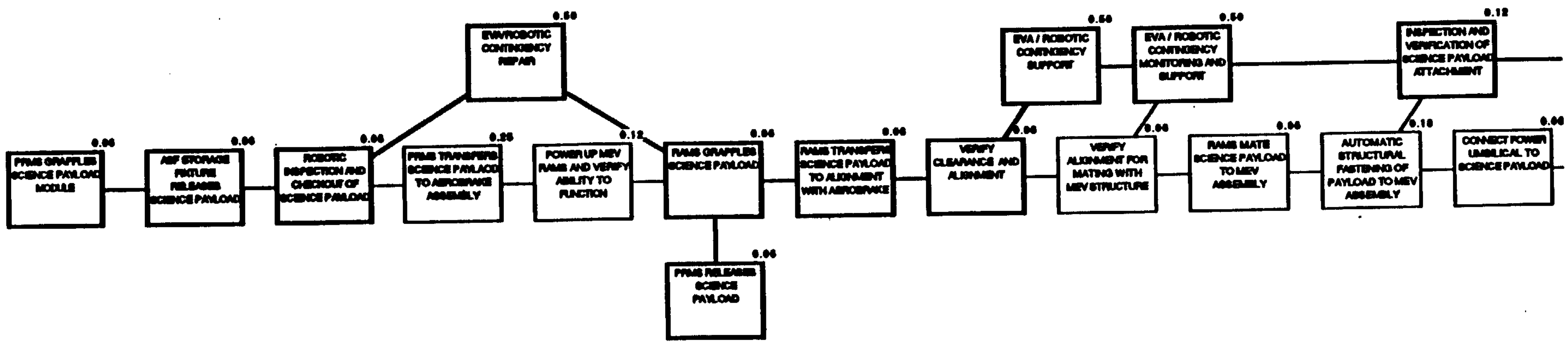

EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.50
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
0.06
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.12
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
READY MEV ASCENT SYSTEM FOR SCIENCE PAYLOAD ASSEMBLY
0.12
MEV ASCENT SYSTEM TO STAND BY MODE
0.12
POWER DOWN AND STORE RAMS
0.06
REMOTE INSPECTION OF FLUID CONNECTIONS
0.12
TEST AND VERIFY FLUID SYSTEMS
0.25

On - Orbit Assembly

The Mars Science Payload installation to the Descent Lander System is shown in the following two pages. Assembly crew monitors operations and performs contingency task(s) as required.

MARS SCIENCE PAYLOAD



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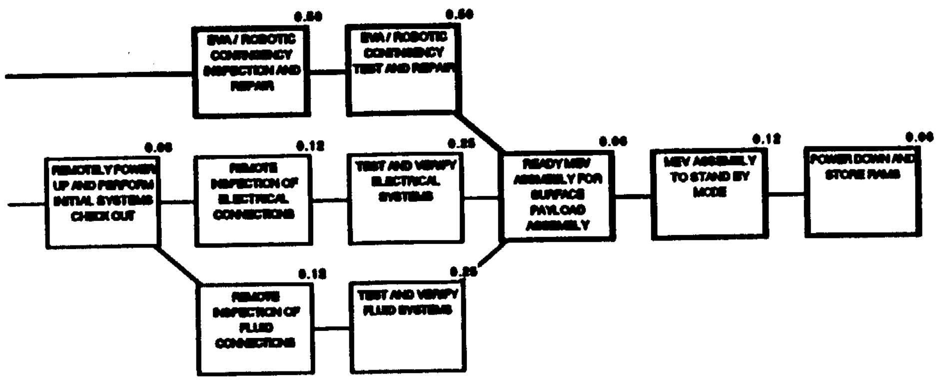
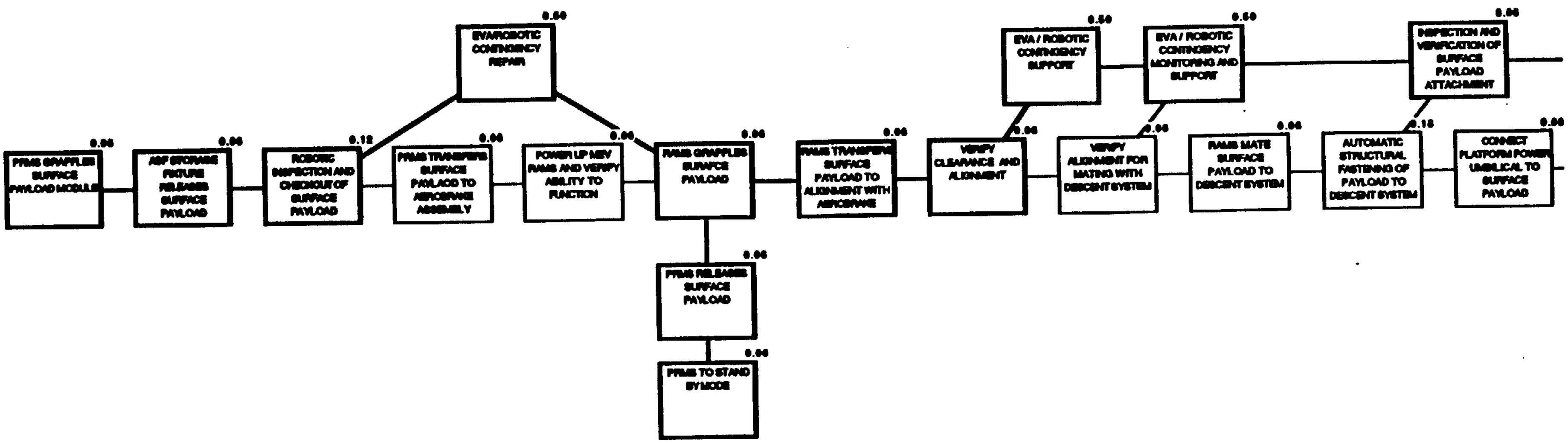

EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
TEST AND VERIFY ELECTRICAL SYSTEMS
REMOTE INSPECTION OF FLUID CONNECTIONS
TEST AND VERIFY FLUID SYSTEMS
READY MEV ASSEMBLY FOR SURFACE PAYLOAD ASSEMBLY
MEV ASSEMBLY TO STAND BY MODE
POWER DOWN AND STORE RAMS

On - Orbit Assembly

The Mars Surface Payload Module installation to the Descent Lander System is shown in the following two pages. The Surface Payload Module is unstowed from its ASF position and transferred to the MEV assembly. Assembly crew monitors operations and performs contingency task(s) as required.

The OMV returns the ACRV and assembly crew to Space Station after the MEV assembly is complete.

MARS SURFACE PAYLOAD



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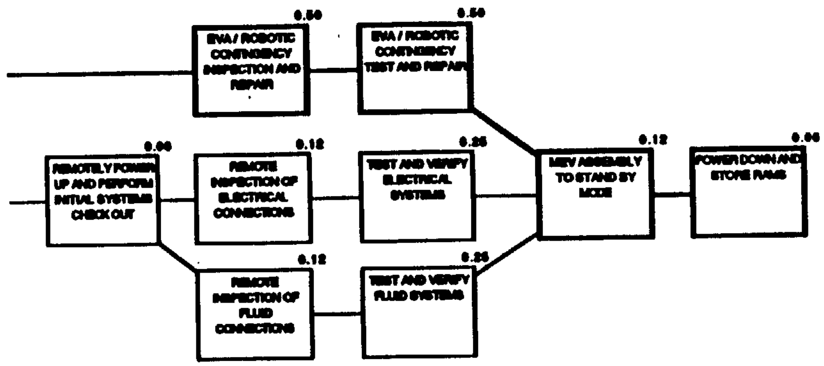

0.50
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.06
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
MEV ASSEMBLY TO STAND BY MODE
0.06
POWER DOWN AND STORE RAMS

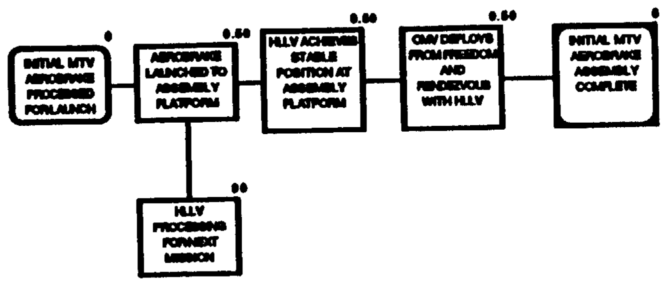
On - Orbit Assembly

The MTV Aerobrake is launched in two (2) HLLV missions, five (5) sections of Aerobrake per mission. The Assembly Sequence for the MTV Aerobrake is shown on the following nine pages.

The RAMS are launched in the first assembly mission to complete the precision mechanical fastening at the joints as required.

The TPS for the joints is installed during the final Aerobrake assembly mission.

HLLV MISSION FIVE



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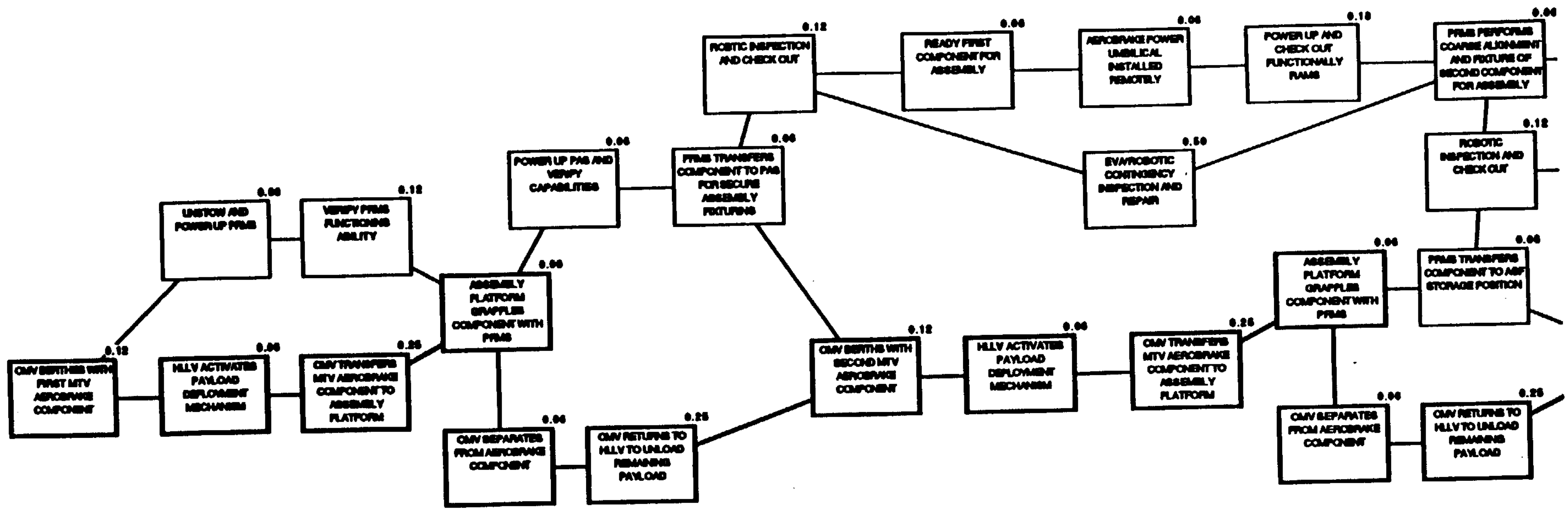
MTV AEROBRAKE MISSION ONE



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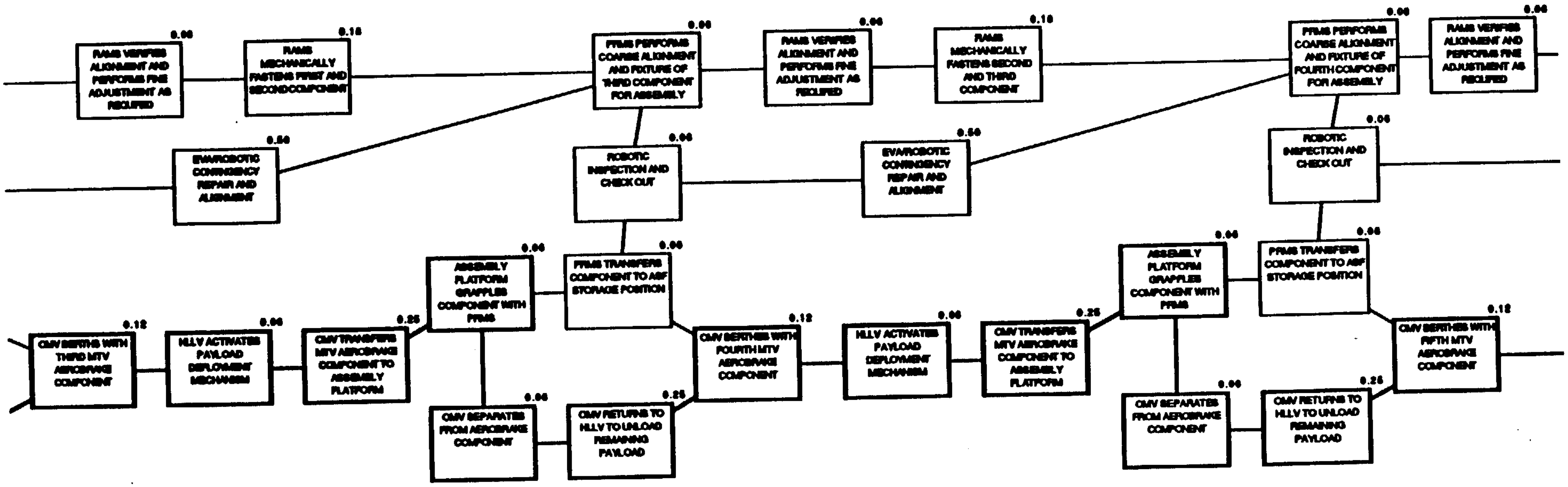

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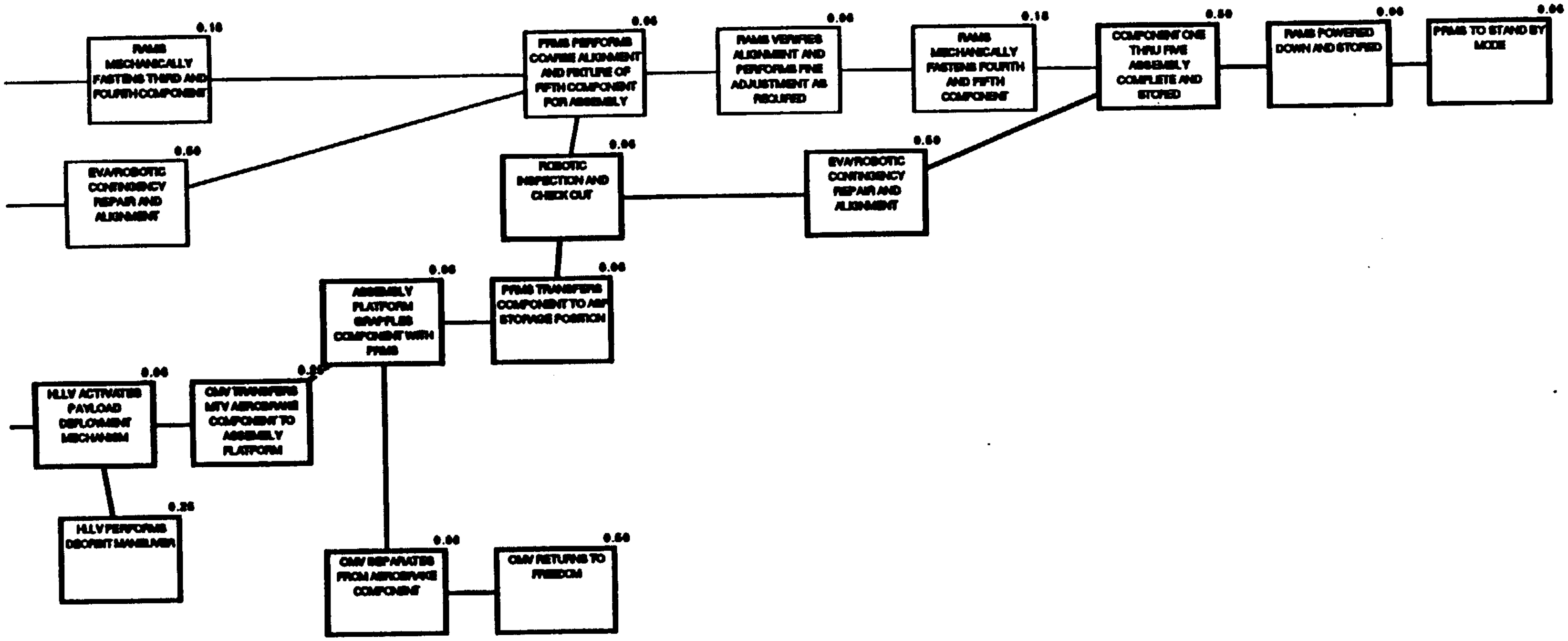

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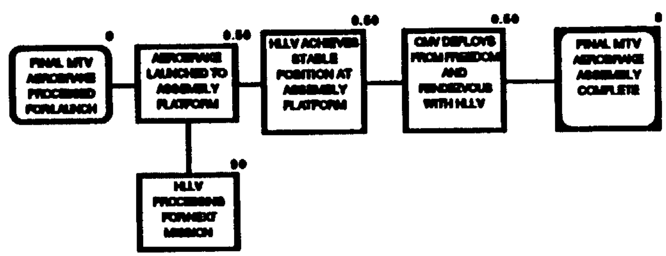
HLLV MISSION SIX



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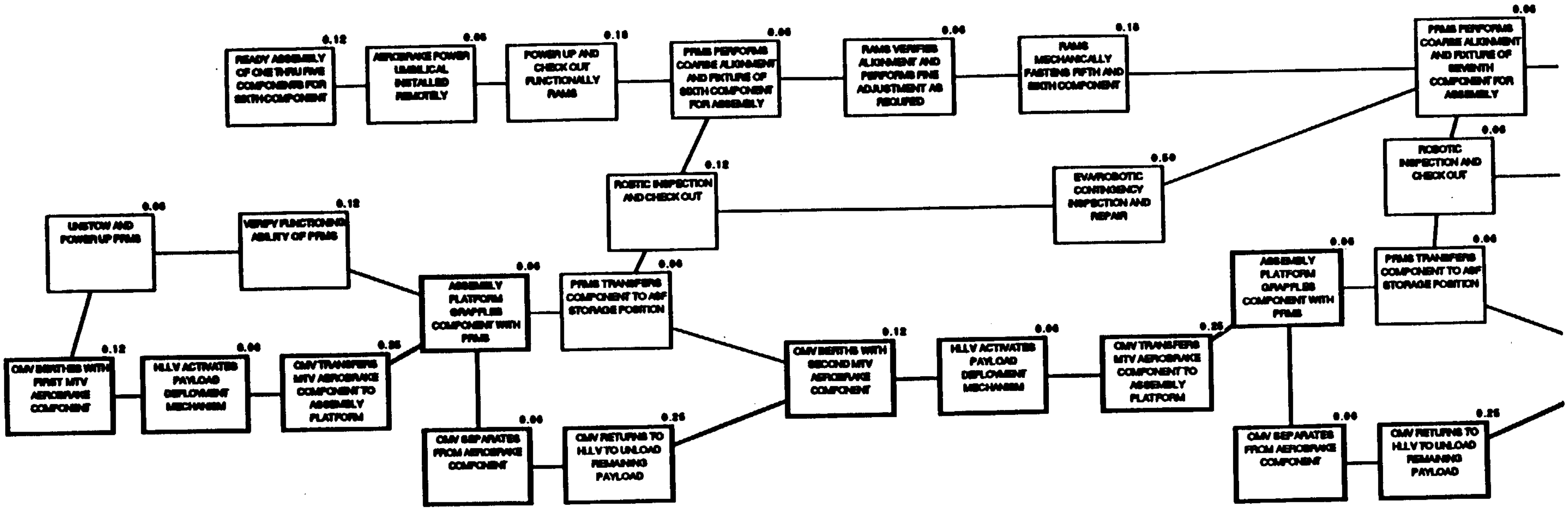
MTV AEROBRAKE MISSION TWO



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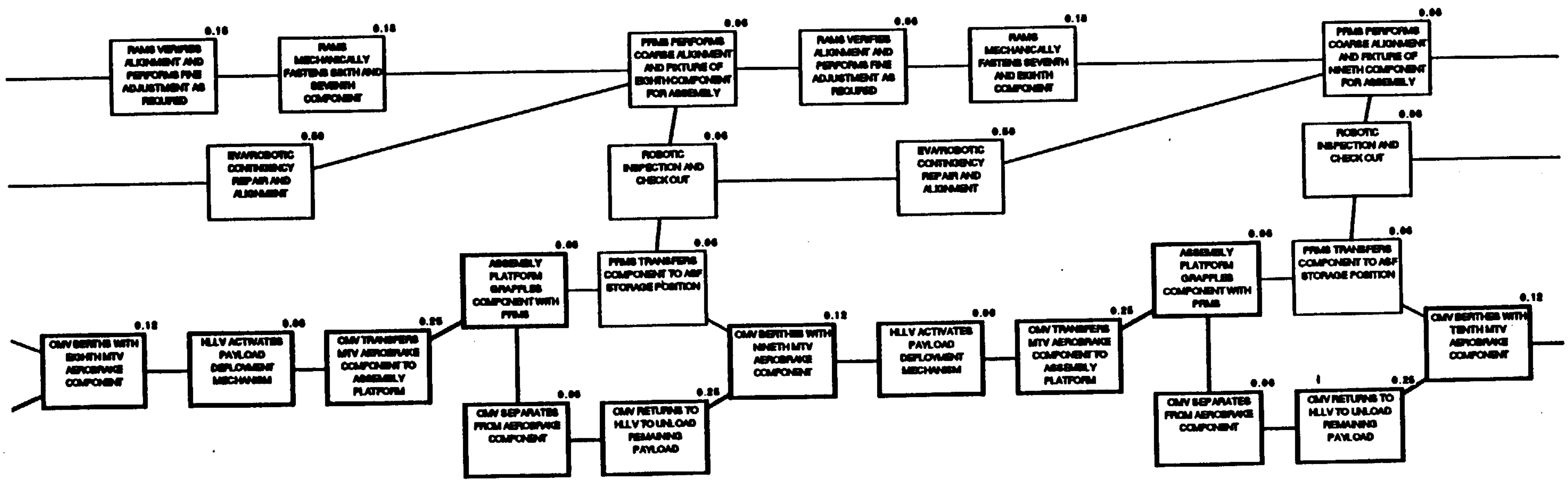
0.18
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS SIXTH AND SEVENTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
FRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF EIGHTH COMPONENT FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS SEVENTH AND EIGHTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
FRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF NINETH COMPONENT FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.12
OMV BERTHS WITH EIGHTH MTV AEROBRAKE COMPONENT
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.06
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH FRMS
0.06
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
FRMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
0.25
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.12
OMV BERTHS WITH NINETH MTV AEROBRAKE COMPONENT
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.06
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH FRMS
0.06
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
FRMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
0.25
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.12
OMV BERTHS WITH TENTH MTV AEROBRAKE COMPONENT

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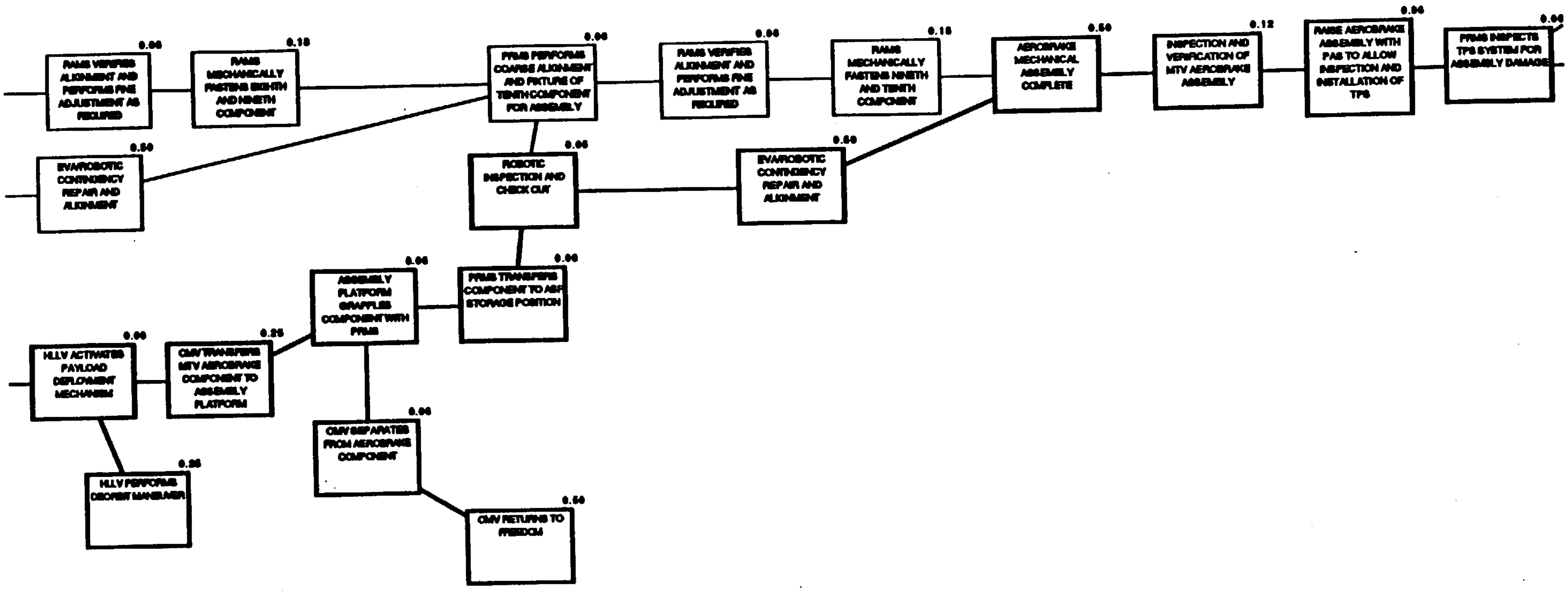
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
RAMS MECHANICALLY FASTENS EIGHTH AND NINETH COMPONENT
0.18
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF TENTH COMPONENT FOR ASSEMBLY
0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
RAMS MECHANICALLY FASTENS NINETH AND TENTH COMPONENT
0.18
AEROBRAKE MECHANICAL ASSEMBLY COMPLETE
0.50
INSPECTION AND VERIFICATION OF MTV AEROBRAKE ASSEMBLY
0.12
RAISE AEROBRAKE ASSEMBLY WITH PAS TO ALLOW INSPECTION AND INSTALLATION OF TPS
0.06
PRMS INSPECTS TPS SYSTEM FOR ASSEMBLY DAMAGE
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.06
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.25
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
OMV RETURNS TO FREEDOM
0.50

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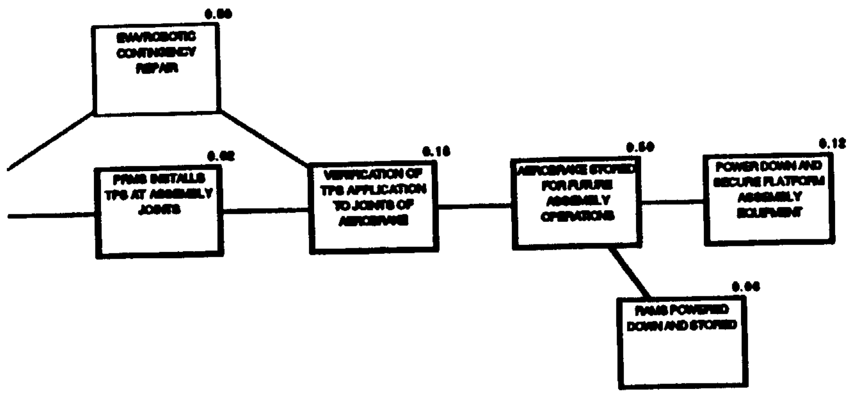

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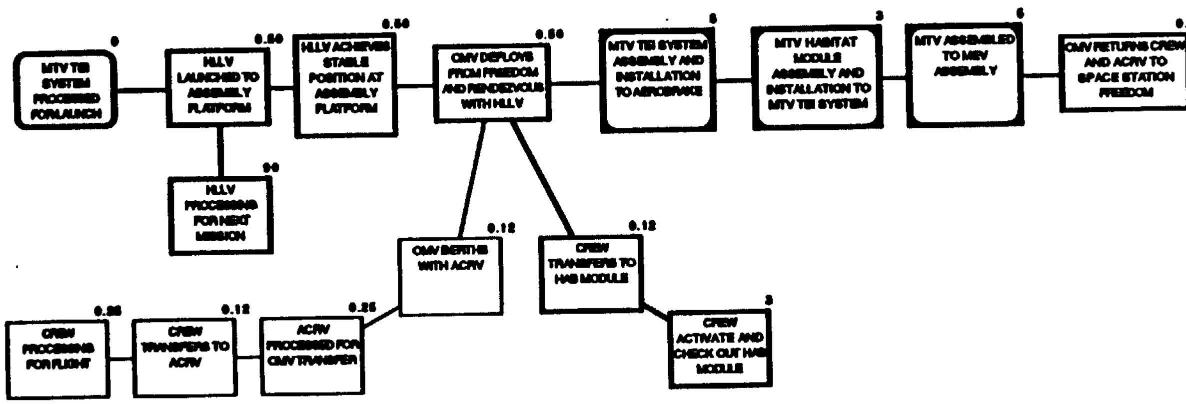
On - Orbit Assembly

The MTV TEI Propulsion System assembly and installation to the MTV Aerobrake is shown in the following three pages.

The OMV transfers the ACRV and Assembly Crew to the MAP. The ACRV is berthed with the Airlock/Node Assembly and the crew transfers to the Habitat Module.

The Assembly Crew monitors operations and performs contingency task(s) as required.

HLLV MISSION SEVEN



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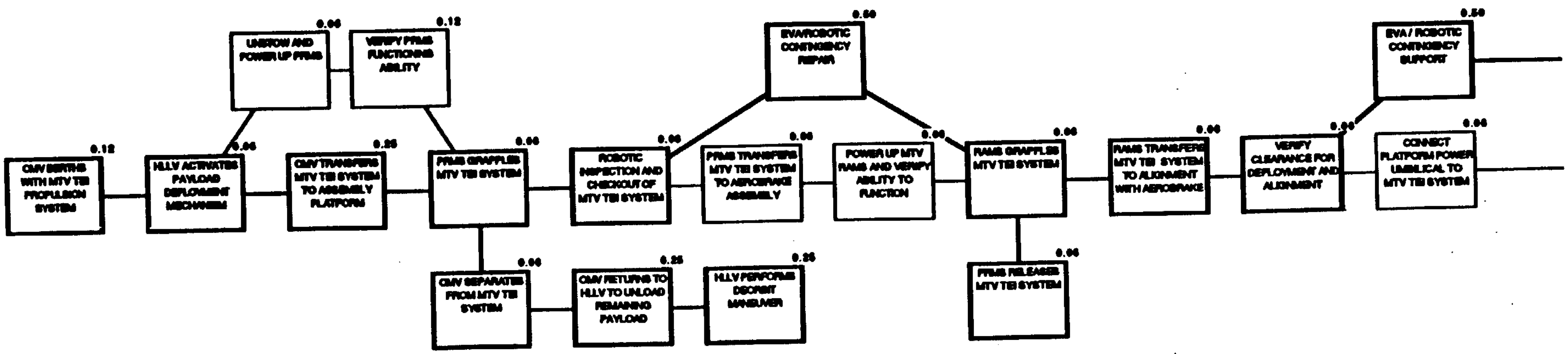
MTV TEI PROPULSION SYSTEM



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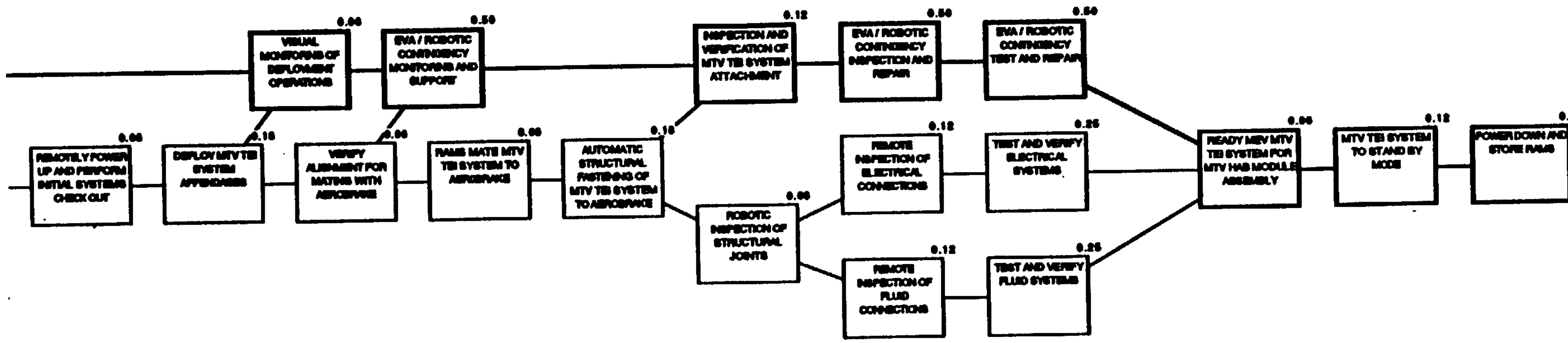

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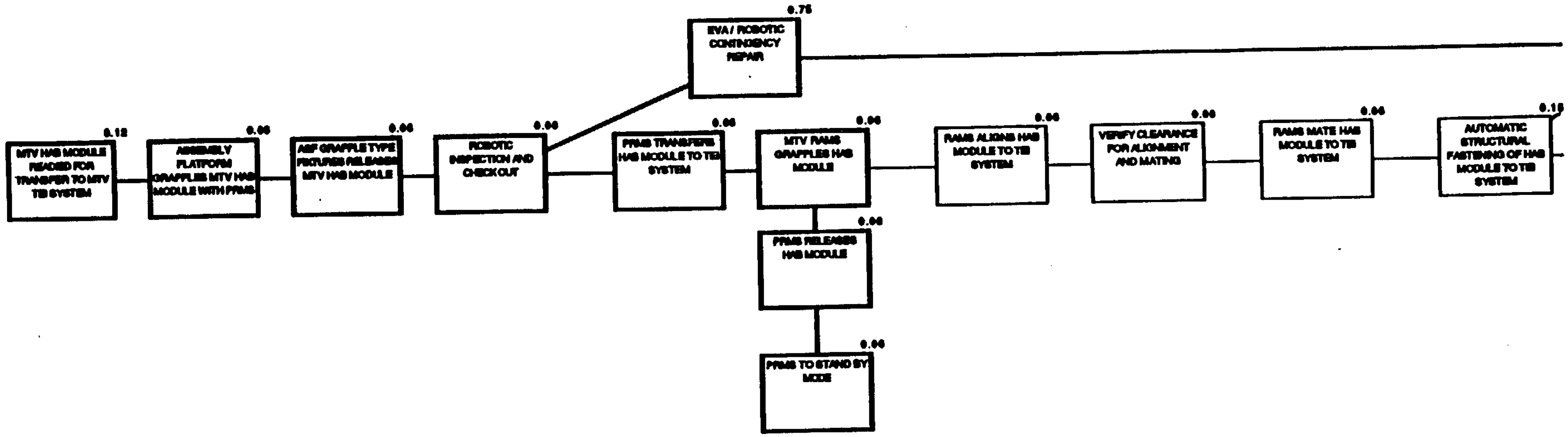
On - Orbit Assembly

The Assembly Crew transfers to the Airlock/Node Assembly and monitors MTV assembly operations.

The MTV Habitat Module assembly and installation to the TEI Propulsion System is shown in the following two pages.

Assembly Crew performs contingency task(s) as required.

MTV ASSEMBLY



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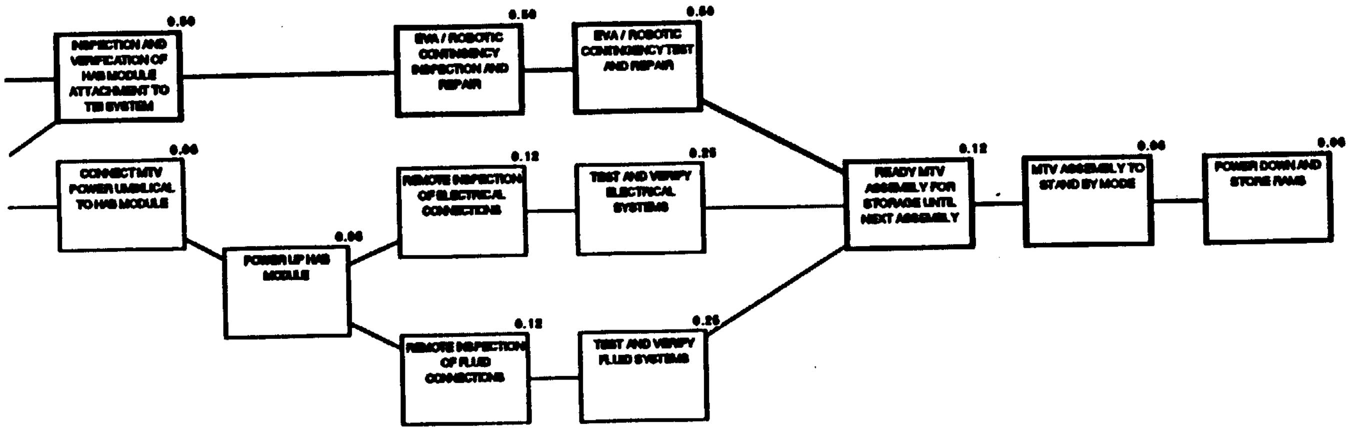

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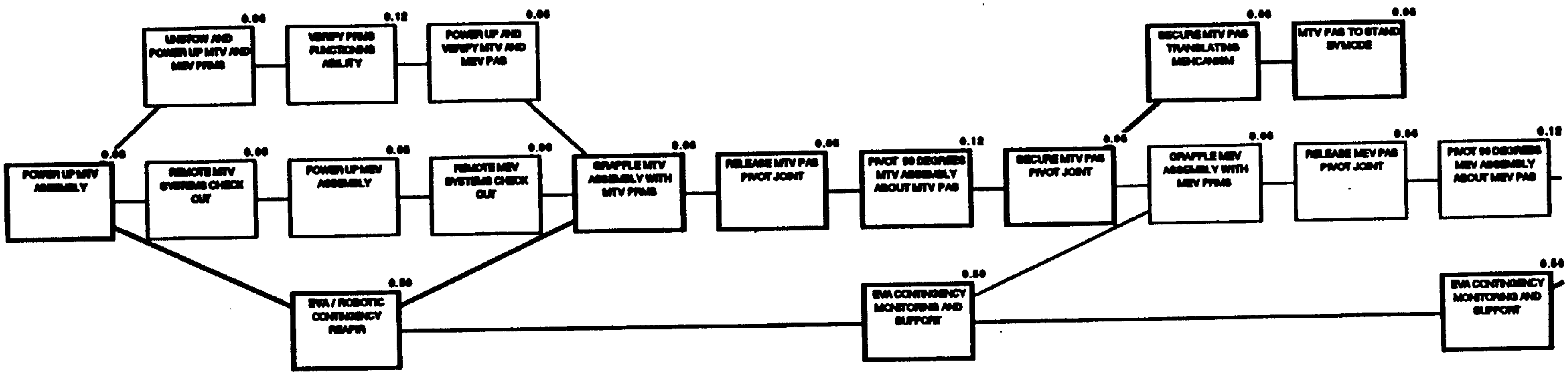
On - Orbit Assembly

The MTV/MEV Assembly is shown in the following two pages.

Assembly Crew monitors operations from the Airlock/Node Assembly and performs contingency task(s) as required.

The OMV transfers the ACRV and Assembly Crew to Space Station after assembly is complete.

MTV / MEV ASSEMBLY



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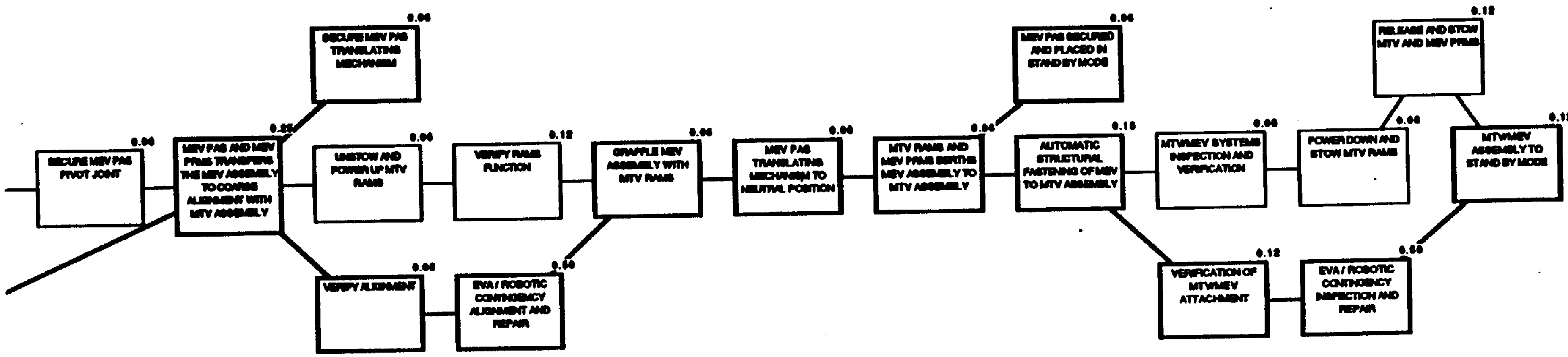

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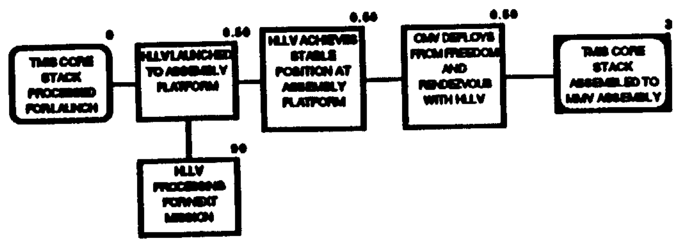
On - Orbit Assembly

The TMIS assembly and installation to the MMV is shown in the following eight pages.

The TMIS core stack is first assembled to the MMV assembly, then the propellant tanks are assembled to the core stack. The assembly sequence for the propellant tanks are identical, therefore, only one example is shown in the following pages.

HLLV MISSION EIGHT



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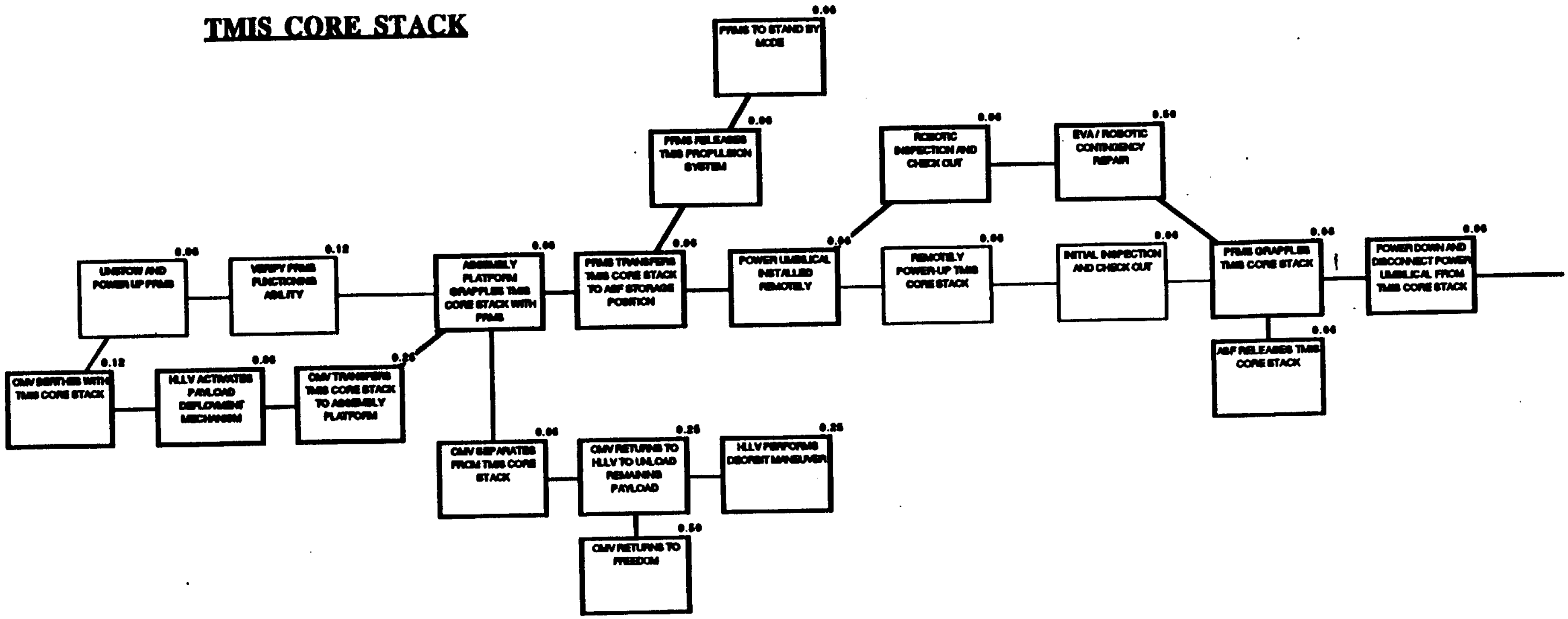
TMIS CORE STACK



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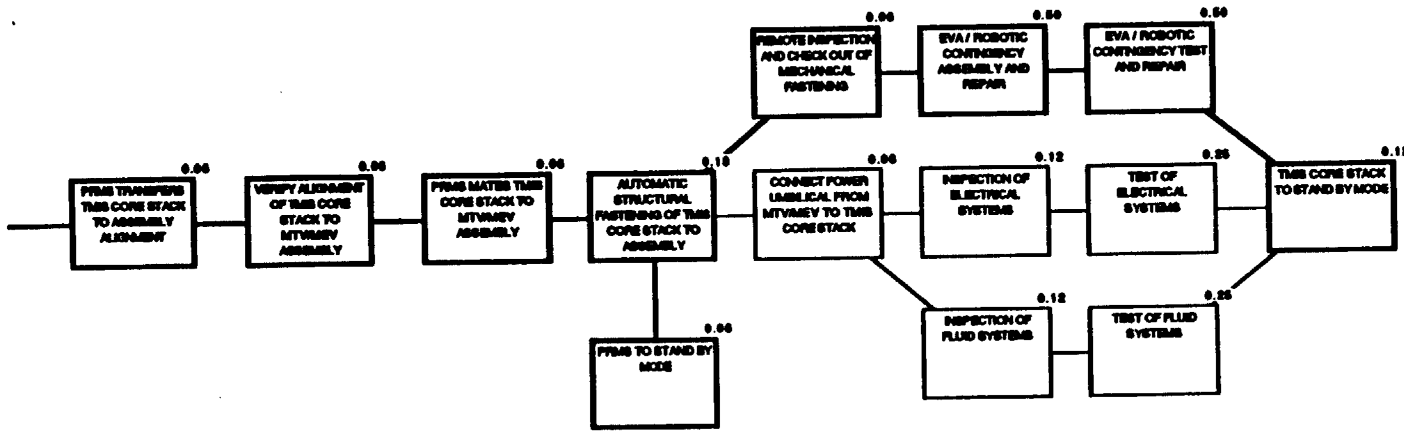

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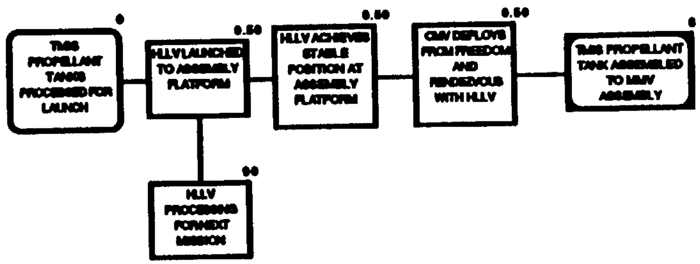
HLLV MISSION NINE



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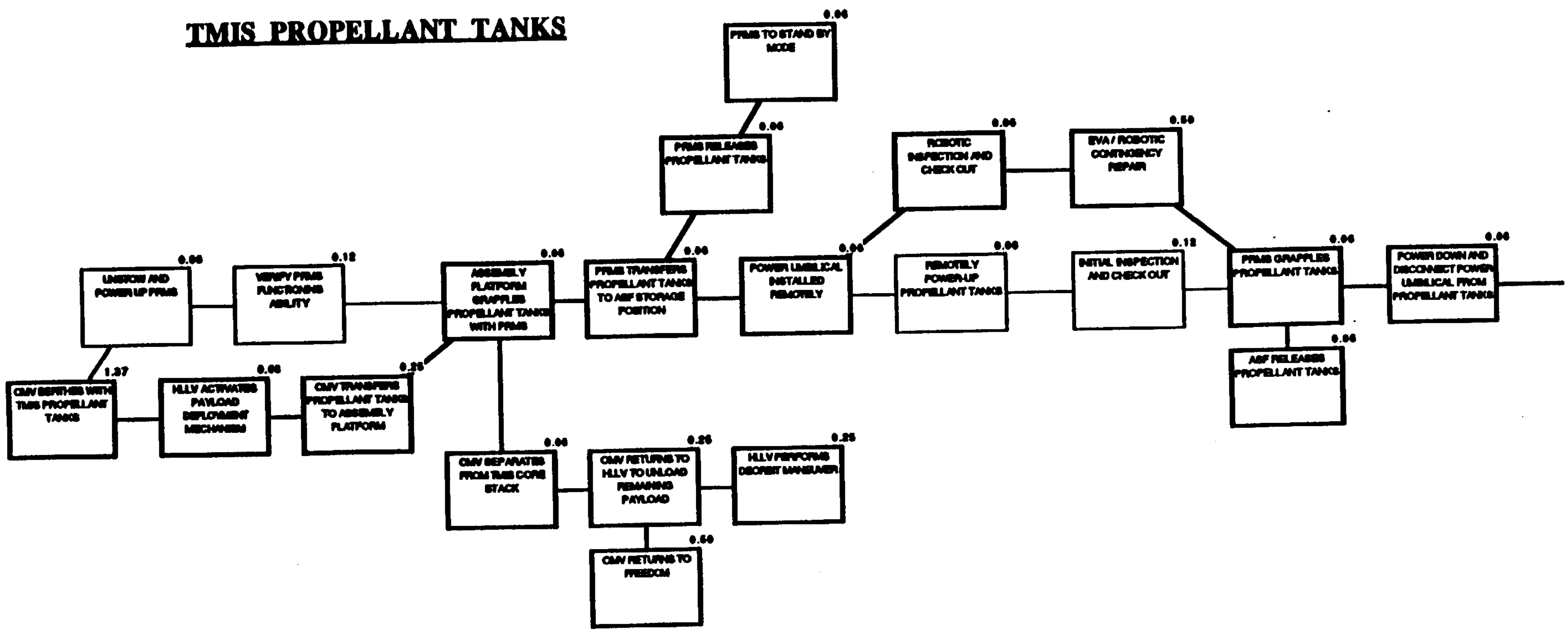
TMIS PROPELLANT TANKS



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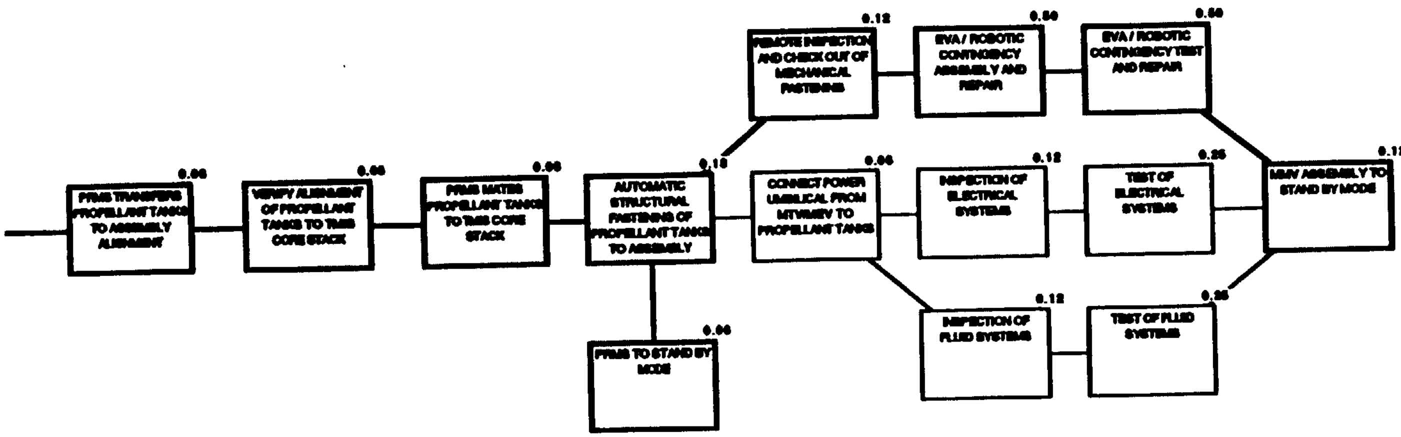

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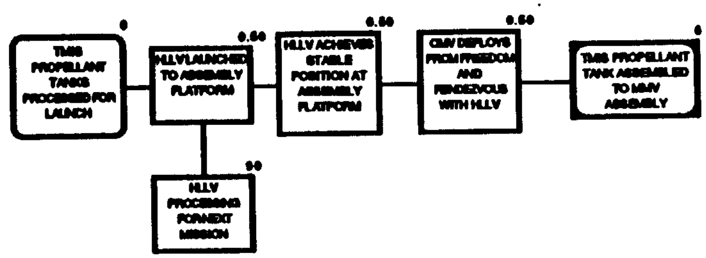
HLLV MISSION TEN



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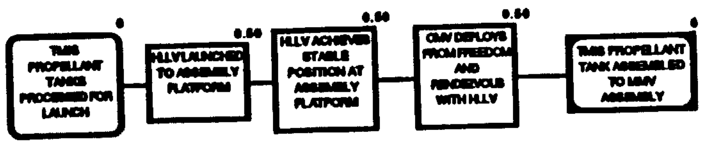
HLLV MISSION ELEVEN



On-Orbit Support Equipment

Assembly scenario is to complete major assemblies robotically with crew support for contingency only.

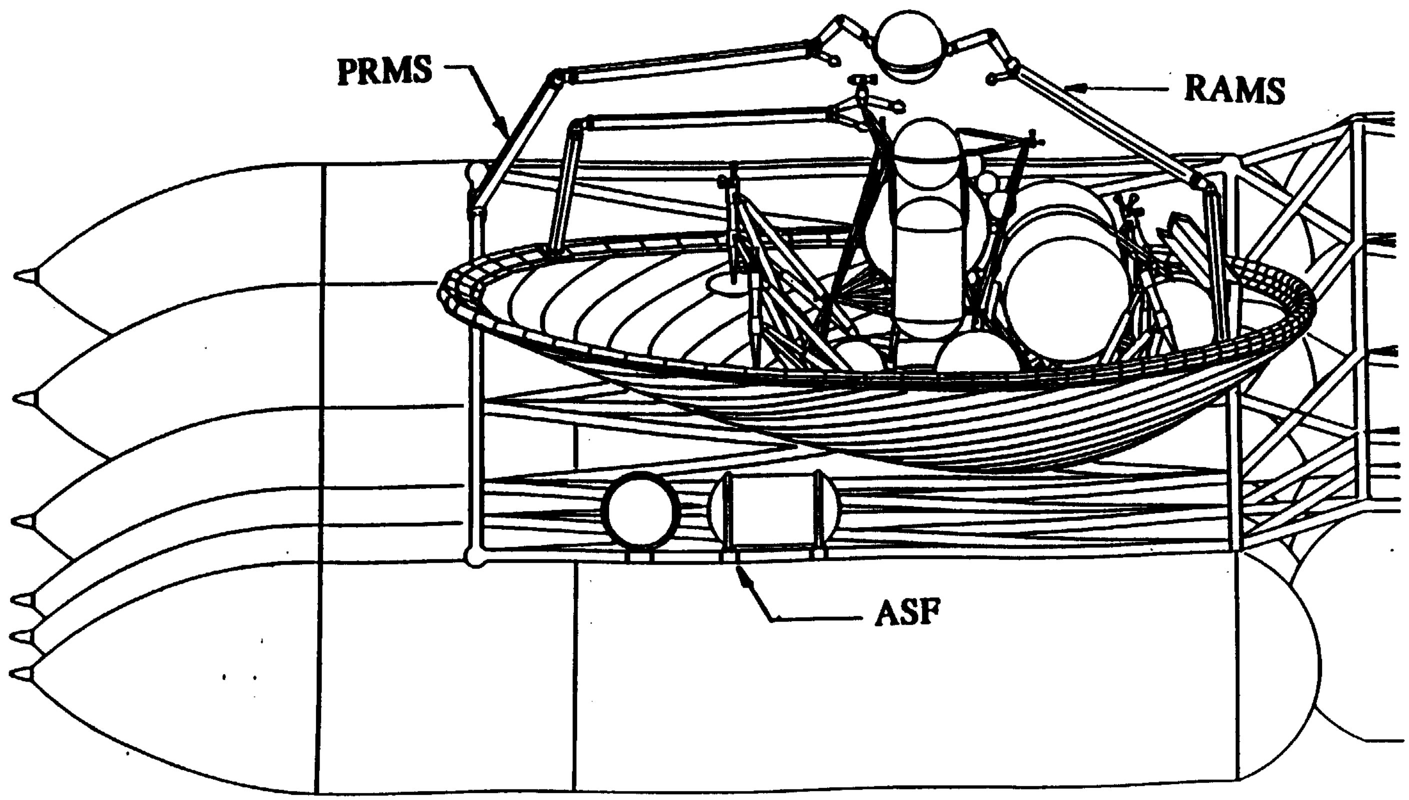
Robotic operations will be controlled by Ground Control Center primarily, with control being "handed off" to assembly crew during contingency operations.

Platform Remote Manipulating System (PRMS)

- Two complete systems required for MMV assembly
- Each main arm can span the entire 30M diameter of the aerobrake
- Each main arm has a "hold down" grapple to secure the working end to EVA handrails
- Each main arm has a 2.5M work arm capable of precise movements and operations
- Elbow joints feature n-pi rotational freedom and the wrist joints are compact roll-pitch-roll units
- Video cameras allow direct monitoring and machine vision from the end effector
- All hardware is bar-coded for positive machine recognition
- End effector is equipped with a 6-axis EM antennae, which determine location and orientation relative to EM beacons distributed across the assembly site.
- Tools and hardware required for assembly operations will be secured to the main arm, within reaching distance of the work area
- Each arm will be capable of maneuvering 128 metric tons (proposed mobile servicing center 10-12-89)
- Each arm will be track-mounted so as to maneuver about the perimeter of the assembly area

On-Orbit Support Equipment

PRMS transfers component to RAMS



MEV Robotic Assembly

On-Orbit Support Equipment (cont'd)

Remote Aerobrake Manipulating System (RAMS)
- Same characteristics as PRMS for commonality
- 2 systems attached to track on each aerobrake

Platform Anchor System (PAS)
- 4 units required for assembly
- Track-mounted to allow movement during assembly operations
- Extendable to TBD height to allow access for TPS installation and inspection
- Lift capabilities of 128 metric tons
- Grapple-type end effector to anchor components to platform
- Elbow joints with 0.50pi rotational freedom
- Wrist joints with roll-pitch-roll movements

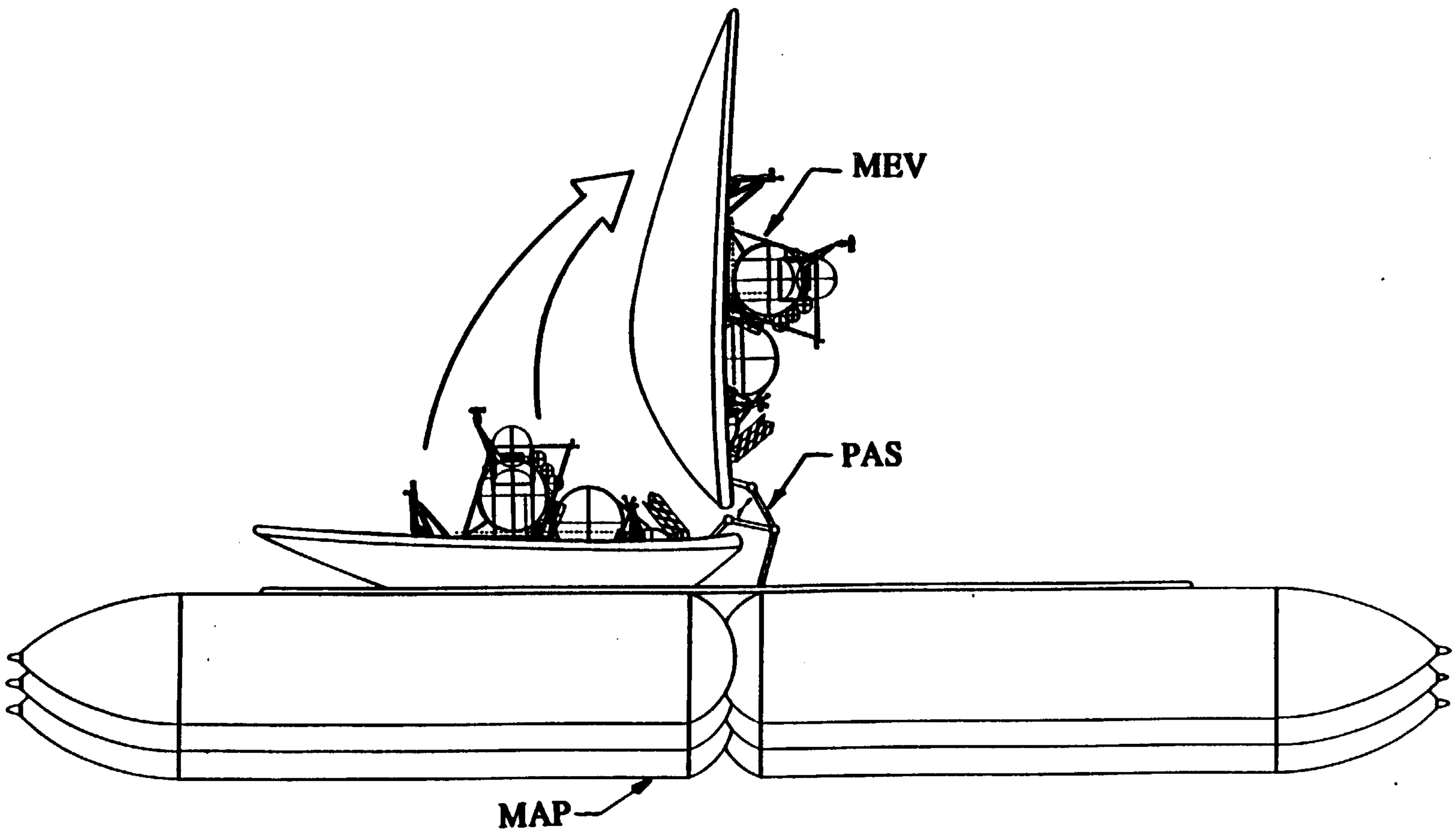
Assembly Support Fixture (ASF)
- Fixed storage locations. TBD units required for assembly across the assembly platform
- Removable grapple-type end effector
- Able to support up to 128 metric tons
- Grapple fitting remotely controlled to release and secure components

Lighting & Video Monitoring
- PRMS and RAMS assembly arms will have required lighting and video/fiber optic monitoring capabilities.
- Portable lighting will be available as required

EVA Handrails and Tether Tie-Down Points
- Available on each assembly component

On-Orbit Support Equipment

PAS lifts and rotates MEV to next assembly position



MEV Robotic Assembly

On-Orbit Support Equipment (cont'd)

Electrical Power
- Will be supplied by assembly platform

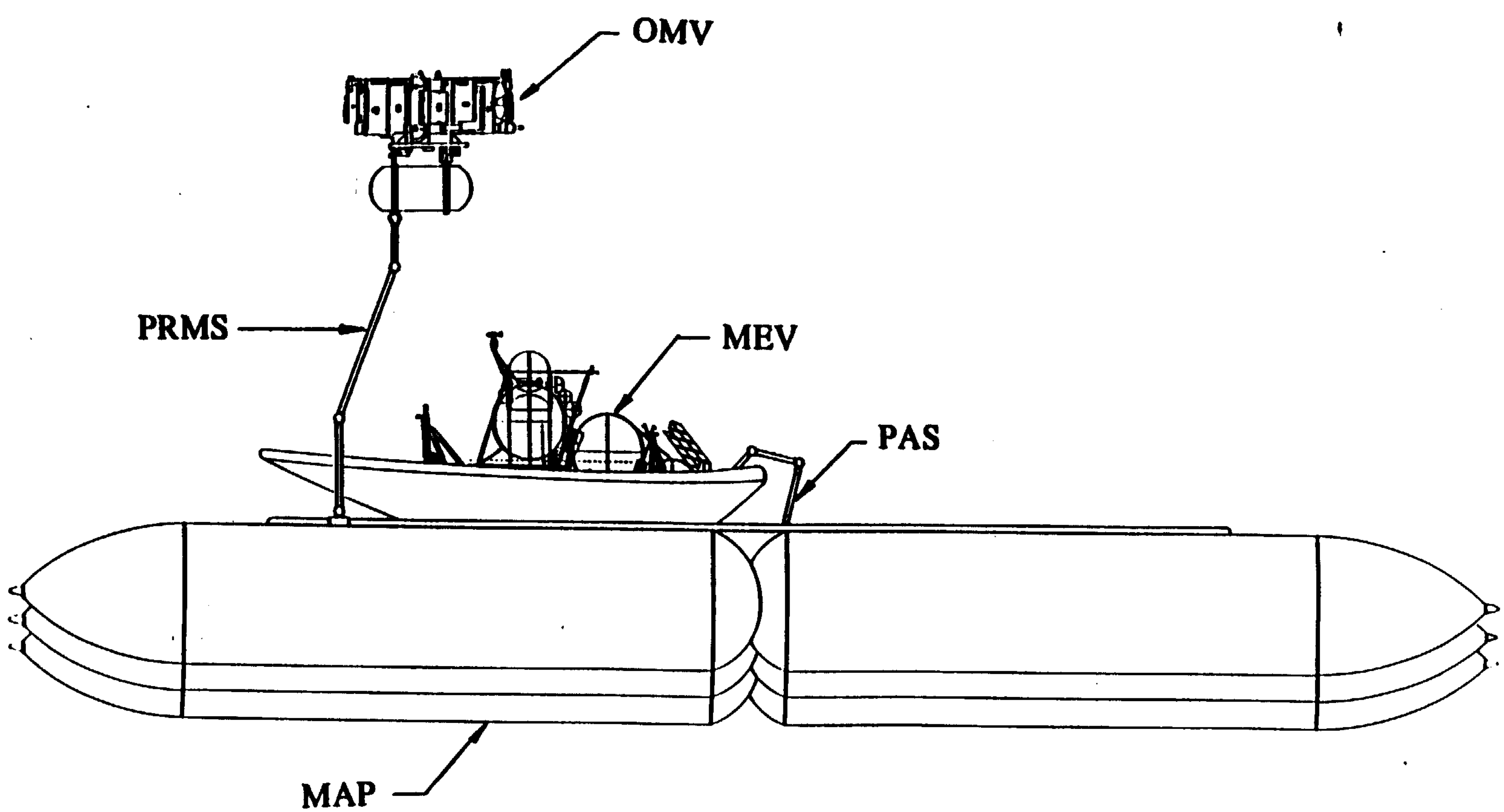
MTV Habitat Module
- Provide crew stationing facility

SSF Type Node
- Houses local control of MAP and Assembly Equipment
- Provide berthing port for Logistics Module
- Provide berthing port for ACRV

SSF-Type Logistics Module
- Provide consumables storage and transportation

On-Orbit Support Equipment

OMV transfers component to PRMS



MEV Robotic Assembly

On - Orbit Assembly Summary

This Analysis shows that On-Orbit Assembly Operations can be established such that EVA Crew time is minimum. Through development of support equipment as defined by this analysis, Crew Operations can be limited to Contingency Assembly and Rerpair, Critical Assembly Monitoring (such as MEV Assembly), and Internal Subsystem Checkout.

On-Orbit Assembly analysis developed facility requirements for an Assembly Node separate from Space Station in LEO. The Assembly Operations will require Debris Protection from the Node. This structure will need Power, Guidance, Attitude Control and Reboost to maintain the self-supported Assembly Node. As defined by the analysis, a Resource Node / Airlock Assembly is required to man-tend the Assembly Node. Other components such as Pressurized Logistics and ACRVs are required to transport the Assembly Crew and Comsumables. For crew stationing and quarters during Assembly Operations, this analysis chose to utilize the MTV Habitat Module.

The requirements for Support Equipment evolved from the previous "Engineering Analysis for Assembly and Checkout of Space Transportation Vehicles in Orbit" (contract NAS2 - 12108) study which began defining Robotic Equipment. This study progressed further by defining exact Robotic Task(s) to complete specific operations. To minimize EVA crew Assembly, development of second generation Robotic Systems (such as Mobile Servicing Center) is crucial to the success On-Oribt Assembly Operations. This analysis is a preliminary definition of On-Orbit Assembly Operations and the Equipment required for Support.

On-Orbit Assembly Summary

- **Crew Interfaces**
  - Internal Subsystem Checkout
  - Critical Assembly Monitoring
  - Contingency Assembly and Repair

- **Facility Requirements**
  - MMV Assembly Platform
  - Power, Guidance, Attitude Control, Reboost, and Debris Protection
  - SSF type Node
  - Airlock
  - SSF type Pressurized and/or Unpressurized Logistic Module
  - MTV Habitat Module - crew quarters

- **Support Equipment**
  - Platform Remote Manipulator System
  - Remote Aerobrake Manipulator System
  - Platform Anchor System
  - Assembly Support Fixture
  - Lights and Video Monitoring Equipment

Issues and Concerns

The TPS is installed on the Aerobrake sections prior to Launch. TPS installation is required at the assembly joints. Method of installation and inspection of TPS on-orbit is a concern.

Payload packages vary in size and shape. The design of payload packages to accommodate shroud diameter and to minimize assembly assembly operations is a concern for Launch Vehicle Intergration.

The Assembly Analysis was structured around Robotic Operations being the primary assembly method. The concern is the high level of robotic intelligency required to perform the assembly task.

Issues and Concerns

- TPS Installation and Inspection
- Launch Vehicle Integration
- Robotic Operations

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Technology

Jerry McGhee

ADVANCED
CIVIL
SPACE
SYSTEMS

Agenda

Technology Development Concerns
High Leverage Technology Issues
Technology vs. Mission Architecture
Continuing Work

Critical Lunar/Mars Reference Technology Development Concerns

A preliminary set of critical technology development concerns was constructed for the Lunar/Mars reference missions. Its purpose is to show a top level representation of the areas which could prove enabling for the reference Lunar and/or Mars missions, without further concentrated research and development, flight testing, and/or precursor missions. Aerobraking may prove enabling for most Lunar and Mars missions, and significantly enhancing for the rest, primarily due to reduced demands on limited Earth to orbit launch capability and lower launch costs. Aeroheating prediction codes cannot be validated without further experimental data (flight or ground simulation data). The degree of development needed for aerobrake TPS materials will be determined by these predictions. Low gravity human factors, to be evaluated on SSF, may affect vehicle design significantly. Foe example, vehicle designs must accommodate artificial - gravity until a need level can be determined from space station based research. Finally, precise mission design, incorperating advanced tracking, telemetry, and GN&C must be verified to accommodate aerobraking and automated rendezvous & docking requirements.

Critical Lunar/Mars Reference Technology Development Concerns

Technology	Comments
High Energy Aerobraking - Thermal protection - High performance structure - Theoretical code validation - Deep space tracking, telemetry, and communication	Heating rates much greater than seen by AFE for Mars capture and Mars - Earth return. High temperature reradiative or lightweight ablative materials must be developed. Precursor missions needed to validate data bases for existing aeroheating/GN&C codes. 17 minute Mars/Earth communications delay will dictate a completely internal GN&C system.
Advanced Space Engine Development - Large engine (150 - 200 klb thrust) - Small engine (15 - 30 klb thrust; throttleable)	High thrust, high Isp cryogenic engine for TMI stage. Low thrust, high Isp, throttleable engine for Lunar/Mars descent and ascent.
Low - g Human Factors	Vehicle designs should accommodate artificial-g configuration until Space Station Freedom based life sciences research can be carried out.
Autonomous System Health Monitoring	Reliable autonomous systems with self monitoring, diagnostic, and correcting capability.
Long Term Cryogenic Storage and Management	Advances in long term low - g cryogenic fluid storage and management required for Lunar/Mars initiatives. Reliable low - g propellant acquisition enabling for all cryo propulsion missions.
Long Duration, High Degree of Closure ECLSS	Reliable SSF validated ECLSS equipment critical for early long term missions.
Efficient Radiation Storm Shelter Material & Configuration	Solar flare prediction / detection capabilities, along with storm shelter designs incorporating effective lightweight materials. Reliable radiation dosimetry techniques are also important.
In - Space Assembly; AR & D	Large aerobraked Lunar and Mars vehicles will require large degree of in - space assembly. AR&D critical for both Lunar/Mars orbital operations.

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

A preliminary set of high leverage technologies was assembled for the Lunar/Mars reference missions. These technologies are enhancing for most, and in some cases, all identified mission architectures. Aerobraking will be significantly enhancing for all Lunar and Mars missions where it is not identified as enabling. Other aerobraking issues which could prove enhancing are lightweight reradiative or ablative TPS material, and ECCV vs. aerocapture of MTV at Earth. Low - g propellant handling and low boiloff cryogenic storage are also very enhancing for any missions where it is not enabling. Advanced propulsion options such as NTR, GCR, SEP, and NEP may prove to be high leverage technology options to baseline cryogenic propulsion systems. Finally, developments in advanced materials can be significantly enhancing in a variety of areas.

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Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

BOEING

Technology	Comments
Aerobraking - Mars Capture (vs. propulsive cap.)	Aerocapture at Mars can reduce IMLEO by more than 50% over propulsive capture.
Aerobraking - Earth Capture (vs. ECCV)	ECCV for Earth return reduces IMLEO and thermal protection system (TPS) requirements. Reusable MTV can reduce life cycle cost.
Aeroshell TPS (reradiative vs. ablative)	Reusable aeroshell requires reradiative TPS at Mars (or thick lightweight ablator), and ablative at Earth. Further advances in materials and processes or mission design may allow for a reradiative Earth/Mars TPS.
Advanced Long Term Cryogenic Storage Technology	Cryogenic boiloff reduction technologies such as advanced MLI design and application, vapor cooled shields, para to ortho H2 conversion, and thermal disconnect support struts can reduce IMLEO significantly with low R & D effort level. Longer missions profit more with increased boiloff to TPS ratio.
Low - g Propellant Transfer	Low - g propellant transfer technology enhancing for all Lunar/Mars mission architectures, and enabling for some Lunar missions.
Efficient Cryogenic Refrigeration System	Cryogenic refrigeration system can reduce vehicle mass and enhance system reliability at the expense of an increased vehicle power level.
O2 - H2 ACS / RCS	O2 - H2 ACS/RCS (Isp = 400 s) reduces system mass considerably over lower Isp storable systems.
High Isp Advanced Space Engine	High Isp advanced space engine (Isp = 485 s) enhances all mission phases for both Lunar/Mars initiatives.
NTR Propulsion System	NTR propulsion system for the trans Mars injection, Lunar transfer, and Mars transfer stages
Advanced In - Space Assembly Techniques	Launch vehicle capability drives on - orbit assembly level. Degree of on - orbit assembly capability affects vehicle configuration, ground assembly/processing, and launch manifesting.
Advanced Materials Development	Advanced materials such as metal and organic matrix composites reduce system inert mass, strength, and/or manufacturing costs. Some advanced materials and processes may prove enabling for some mission architectures. (ex: Mars/Earth capture aerobrake

Required Technologies vs. Alternative Mission Architecture

A set of required technologies for the seven identified alternative mission architectures outlined in the evolotionary concepts section is presented. The purpose of this matrix is to provide a preliminary comparison of technology development needs for the alternative architectures. The matrix also serves to better define the architectures. From this top level matrix, a more detailed set of technology requirements can be derived. A set of accommodating technologies can be compiled for needs areas where options exist. Finally, the technology areas can be prioritized as enabling and enhancing, and a return on investment performed for identified high leverage technologies. This portion of the matrix includes most of the cryogenic management issues. Enabling technologies are represented by the filled circle, and enhancing technologies by the open circle. Extensive low - g cryogenic propellant launch, acquisition and transfer refers to the Mars conjunction case, and the mass driver option, where propellant will be used for the transfer vehicles, which will be parked in a low - g environment (Lunar or Mars orbit, or libration staging point). The Mars cycler orbit case includes a question mark for the long term cryogenic storage system, because the necessary thrust levels and type of propulsion system are undetermined at this time.

ADVANCED CIVIL SPACE SYSTEMS

Required Technologies vs. Alternative Mission Architecture

BOEING

	Low boiloff cryogenic propellant storage system (1-3 yr)	Low boiloff cryogenic propellant storage system (15 - 60 d)	Low - g fluid acquisition and transfer	Extensive low - g cryogenic propellant launch, acquisition, and transfer	Cryogenic tank integrity monitor	Cryo fluid rematible umbilical	Lunar LOX production, liquification, and transfer technology	Mars O2 production, liquification, and transfer technology
Mars NEP Alternative Architecture	○	●	●		●		●	●
Lunar/Mars NTR Alternative Architecture	●	●	●		●		●	
Mars SEP Alternative Architecture	○	●	●		●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●	●	
Mars Cycler Orbits Alternative Architecture	?	●	●		●	●	●	
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●	●	● + H2
Lunar / Mars NEP Alternative Architecture	○	●	●		●		●	●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix section represents the major aerobraking concerns. The aerobraking energy columns for Mars and Earth capture digresses from the format in order to illustrate the energy levels, and therefore, the level of technology development needed for the various architectures. Aeroheating predictions, reusable aerobrake TPS, advanced GN&C, and TT&C follow along with the high and medium energy missions. Again, a question mark is shown for the Mars cycler orbit case. Reusable TPS for Earth return cannot be determined as a technology development concern until the aeroheating load at Mars can be determined for the cycler orbits. Further mission design efforts must be carried out before an estimate on this can be made.

Required Technologies vs. Alternative Mission Architecture (Cont.)

ADVANCED CIVIL SPACE SYSTEMS

BOEING

	Earth return aerobrake energy	Mars capture aerobrake energy	Mars lander aerobrake	High performance aerobrake structure	Aerobrake assembly and test	Aeroheating prediction (Earth and/or Mars)	Reusable aerobrake TPS for Earth return	GN & C to protect TPS	Advanced high accuracy and rate TT & C	In space AR&D / assembly
Mars NEP Alternative Architecture	Low	Low	●	●	●					●
Lunar/Mars NTR Alternative Architecture			●	●	●					●
Mars SEP Alternative Architecture	Low	Low	●	●	●					●
L2 Node / Mass Driver Alternative Architecture	High	High	●	●	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	High	High	●	●	●	●	?	●	●	●
Mars Conjunction/Direct Alternative Architecture	Medium	Medium	●	●	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	Low	Low	●	●	●					●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix area represents the major propulsion issues, with the exception of the radiation protection system, for the baseline and alternative mission architectures. The system to inert and can waste for radiation shielding can be enhancing, while a GCR and ALSPE shelter is enabling for all mission architectures. Again, due to the undefined Mars cycler orbit trajectories, it is questionable as to the need for a large cryogenic space engine. A H_2 - O_2 ACS/RCS system is noted as enabling for each option, as it will be for any option over a baseline storable system. A Lunar orbital momentum storage and transfer device such as a bolo can be enhancing for all missions, after an initial launch and assembly penalty for the massive (~ 1000 Mt) device.

ADVANCED CIVIL SPACE SYSTEMS

Required Technologies vs. Alternative Mission Architecture (Cont.)

BOEING

	Large (150 - 200 klb) cryogenic advanced space engine	Small (15 - 30 klb) cryogenic advanced space engine	H2 - O2 ACS/RCS	Multi - MW space based nuclear electric power	Multi - MW space based nuclear thermal power	Surface nuclear electric power	Multi MW solar power system (arrays and handling equip.)	Radiation protection (system to inert & can waste)	Mass driver / rail gun technology	Lunar orbital momentum transfer device (Bolo)
Mars NEP Alternative Architecture		●	○	●		●		●		○
Lunar/Mars NTR Alternative Architecture		●	○		●	●		●		○
Mars SEP Alternative Architecture		●	○				●	●		○
L2 Node / Mass Driver Alternative Architecture		●	○			●		●	●	○
Mars Cycler Orbits Alternative Architecture	?	●	○			●	●	●		○
Mars Conjunction/Direct Alternative Architecture	●	●	○			●		●		○
Lunar / Mars NEP Alternative Architecture		●	○	●		●		●		○

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

The final section of the matrix is not as illustrative as the others, in that all of the listed technologies are enabling, with the exception of a closed ecological life support system, which is significantly enhancing for all identified mission architectures.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Autonomous health monitoring and check - out	High data rate comm. or high performance compression	DMS/system diagnostics. Art. intell/neural nets/high processing rate GN&C	Long duration refurbishable crew habitat	Long duration BCLSS	CELSS
Mars NEP Alternative Architecture	●	●	●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●	●	●	●
Mars SEP Alternative Architecture	●	●	●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	●	●	●	●	●	●
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	●	●	●	●	●	●

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Technology Task Continuing Work

Current and planned technology tasks to be performed during the next interim period, and the remainder of the contract period are outlined. The main thrust of the work during the next reporting period will be the completion of a concentrated technology assessment, and preliminary technology screening based on its results. Technology assessments and screening will continue throughout the remainder of the contract period, and a return on investment analysis, based on life cycle costs, will be completed for identified high leverage technologies.

Technology Task Continuing Work

Near term technology task work will be concentrated in the following areas:

- Completion of a concentrated technology assessment begun on March 16
- Compilation of a core technology set for the Lunar reference (in addition to the existing Mars reference set), along with available identified technology options.
- Preliminary screening of identified technologies based on information compiled during the concentrated technology assessment. The screening will be performed primarily to evaluate and eliminate technologies by availability vs. need level, and satisfaction of performance requirements.

Continuing technology task work is planned as follows:

- Continuing technology assessments as required, on an individual basis
- Additional screenings based on affect on IMEO, availability, or technical feasibility
- Return on investment analysis for high leverage technologies

Mars Reference Vehicle Technology Requirements

Technology requirements, broken down by stage, are presented in detail for the Mars reference vehicle. This exercise was performed in order to aid in the identification of technology development concerns for the reference mission architecture. Another mission of this work is to provide a complete record of technology systems performance levels derived or assumed in the Mars reference vehicle design. A similar set of technology requirements will be derived for the reference Lunar vehicle. A more detailed set of technology requirements will be contained in the technology assessment database, currently being assembled in the concentrated technology assessment, identified in the technology task continuing work chart.

Mars Reference Vehicle Technology Requirements

I. TMIS

A. Cryogenic storage system

1. Thermal protection system - MLI over foam. (1" foam; ~ 1" MLI)
2. Tanks launched wet.
3. Thermodynamic vent coupled to a single vapor cooled shield.
4. Topoff before Earth departure.
5. ~ 6 months in LEO before use.
6. Negligible boiloff loss after topoff.

B. Propulsion

1. Isp = 475 s
2. Thrust = 150 klb/engine
3. Advanced space engine.
4. Nozzle area ratio = 400
5. No throttling requirements.
6. Gimbal angle (nominal) = 10°
7. Up to 3 burns for departure maneuver (2 restarts).
8. Engine out capability (crossfeed propellant lines).
9. No specified engine cycle.
10. In-space changeout capability.
11. Off vehicle preflight checks.
12. No retraction / extension required.

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Mars Reference Vehicle Technology Requirements (cont.)

C. <u>Structure</u>
1. Material - metal matrix composites, advanced alloys, and organic matrix composites.
2. Meteor/debris protection provided for tanks and plumbing.

D. <u>Avionics</u>
Piggybacked on MTV.

E. <u>Power</u>
1. Level : < 1 kW
2. System: Auxiliary power units on engine pod; piggybacked on MTV for back-up.

F. <u>Assembly</u>
1. Off station assembly.
2. Degree of assembly: Separate tanksets / propulsion modules connected in LEO to form propulsion stage.

II. MTV

A. <u>Cryogenic storage system</u>
1. Thermal protection system - MLI; 100 layers on H_2 & O_2 tanks (2").
2. Tanks launched wet - no transfer other than topoff before Earth departure.
3. Thermodynamic vent coupled to a series of vapor cooled shields on the H_2 tank, and one on the O_2 tank.
4. Topoff in LEO before Earth departure.
5. ~9 months in LEO before Earth departure.
6. Boiloff loss of < 10% before Mars departure.

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Mars Reference Vehicle Technology Requirements (cont.)

B. Propulsion

1. Isp = 475 s.
2. Thrust = 30 klb/engine.
3. Nozzle area ratio = 400.
4. No throttling requirements.
5. Gimbal angle (nominal) = 10°
6. M/D shield for plumbing & tanks.
7. 3 burns @ 4 - 6 month intervals - minimal degradation.
8. 2 restart capability.
9. Engine out capability (crossfeed propellant lines).
10. Expander cycle.
11. In-space change out capability.
12. Off vehicle preflight checks.
13. No retraction/extension required.

C. Structure

1. Vehicle
   a. Metal matrix composites / advanced alloys / organic matrix composites.
   b. Micrometeoroid protection for habitat structure (shell and insulation).

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Mars Reference Vehicle Technology Requirements (cont.)

2. Aerobrake
   - a. L/D = 0.5
   - b. Crossrange: NA
   - c. Vhp = 7.07 km/s.
   - d. Max-g loading = 6.
   - e. Max. temperature = 4000° F.
   - f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   - g. TPS material: Advanced reradiative tiles.
   - h. Relative wind angle (reference) = 20°.

D. <u>Avionics</u>

1. Planetary vicinity -
   - a. Relative velocity error = 100 m/s.
   - b. Relative position error = 25 km.

2. System -
   - a. Relative velocity error = 100 m/s.
   - b. Relative angle error = 0.5°.

E. <u>Power</u>

1. Level - 15 kW.
2. System: Solar arrays with battery storage (NiCad).
3. Back up system: NA

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Mars Reference Vehicle Technology Requirements (cont.)

F. Assembly
1. Off station assembly.
2. Assembly level (complexity): TBD

G. Habitat
1. ECLSS: Space Station Freedom derived system with similar degree of closure; potable H_2O from cabin condensate; CO_2 reduction/regeneration; Hygiene H_2O from urine processing. CELSS to be evaluated.

2. Structure
   a. 2219 - T8 aluminum pressure vessel.
   b. Pressurized to 20 psig on launch for structural integrity.
   c. Insulation & M/D shield external to pressure shell.
   d. No penetrations in end domes.
   e. Radiation storm shelter provided, and configured to utilize equipment & supplies as partial shielding.
   f. External space radiator integral with M/D shield.

3. Cabin repressurizations: 2+ (outbound emergency could use propellant for repress.)
4. Spares: 15% of active equipment - component level.
5. Redundancy: Two complete and separate systems for life critical systems + spares. Component changeout capability.
6. Residence time = 535 days.
7. Science: Transit science as allowed by individual mission.
8. EVA capability: EVA suits provided for all crew; EVA waste fluid recovery for ECLSS.

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Mars Reference Vehicle Technology Requirements (cont.)

H. ECCV

1. Apollo size & style.
2. Open ECLSS (LiOH, no H_2O recovery).
3. Residence time: 2 - 3 days.
4. Propulsion: RCS only.

III. MEV

A. Cryogenic storage system

1. Thermal protection system: 100 layers of MLI for H2 and O2 tanks (2").
2. Tanks: double wall tanks with vacuum annulus;
   low thermal conductivity support system for inner tank.
3. Thermodynamic vent: Simple design for gravity field.
4. Tanks launched dry and filled prior to descent, from MTV tanks, or refrigerated. (no boiloff prior to descent)
5. Stay time from 30 - 600 days on Mars surface.
6. Boiloff level < 20% for surface stay.

B. Propulsion

1. Isp = 460 sec.
2. Thrust = 30 klb / engine.
3. Nozzle area ratio = 200.
4. Throttleability = 15:1.

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Mars Reference Vehicle Technology Requirements (cont.)

B. Propulsion (cont.)
6. Gimbal angle (nominal) = 10°.
7. No restart capability necessary for nominal case.
8. Space storage time between burns : NA.
9. Engine out capability (crossfeed propellant lines).
10. Expander cycle.
11. In-space changeout capability.
12. Off vehicle preflight checks.
13. Retraction / extension capability.

C. Structure
1. Vehicle
   a. metal matrix composites / advanced alloys / organic matrix composites.
   b. Micrometeoroid protection for tanks and plumbing.

2. Aerobrake
   a. L/D = 0.5 to 1.0
   b. Crossrange: 1000 km.
   c. Vhp = 7.07 km/sec.
   d. Maximum g loading: 6.
   e. Maximum temp: TBD (estimated 3100° F).
   f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   g. TPS material: Advanced reradiative tiles.
   h. Relative wind angle (reference) = 20°.

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Mars Reference Vehicle Technology Requirements (cont.)

D. Avionics
   1. Error without beacon = 1 km.
   2. Touchdown error = 1 m/s.
   3. Obstacle avoidance capability.

E. Power
   1. Level: ~ 2.5 kW.
   2. System: fuel cells (regenerable).
   3. Back-up system: abort to orbit.

F. Assembly
   1. Off station assembly.
   2. Assembly level (complexity): TBD

G. Habitat
   1. ECLSS: open system; stored potable H_2O; LiOH CO_2 adsorption.
   2. Structure
      a. Aluminum (2219 - T8) pressure vessel.
      b. Overpressurized on launch for structural integrity.
      c. Insulation and micrometeoroid protection external to pressure vessel.
      d. No penetrations in end domes.
      e. No radiation shelter provided in MEV.
      f. External space radiator integral with micrometeoroid shield.
   3. Repressurizations: 2.
   4. Spares: 15% of active equipment mass; component level.
   5. Redundancy: EVA suits as backup to cabin repressurization.; no system level ECLSS redundancy required due to low complexity open system.
   6. Residence time: ~3 days (surface systems support surface stay).
   7. Science: none.
   8. EVA capability: provided for all crew; transferred from MTV.

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Piloted Rover

ADVANCED CIVIL SPACE SYSTEMS

Agenda

Piloted Rovers Task

Piloted Rovers Technology Needs Study

Piloted Rovers Technology Needs Study Program Master Flow

This chart presents the schedule for the Piloted Rovers Technical Program. At the present time, only the first phase of eight (8) months is funded with the second phase of ten (10) months tentatively scheduled.

During the first four (4) months, the emphasis was on establishing a state-of-the-art survey, obtaining mission data from centers involved in mission definitions and in establishing performance requirements for the set of vehicles determined to be required during each task phase.

During the second four months, the emphasis will be placed on continuation of the state of the art survey, defining vehicle configurations for each of the mission tasks, establishing evaluation measures for selecting vehicle configurations and subsystem concepts. Trade studies also need to be conducted in order to select more promising concepts and common subsystems for the vehicle. All this will be documented in the Phase 01 report.

Phase 2 is essentially to be a repeat in more detail of the Phase 01 effort focusing on the concepts found to have the most promise. More emphasis will also be placed on the astronaut tasks to be performed and in alternate uses of the vehicles to aid in these tasks.

Piloted Rovers Technology Needs Study
Program Master Flow

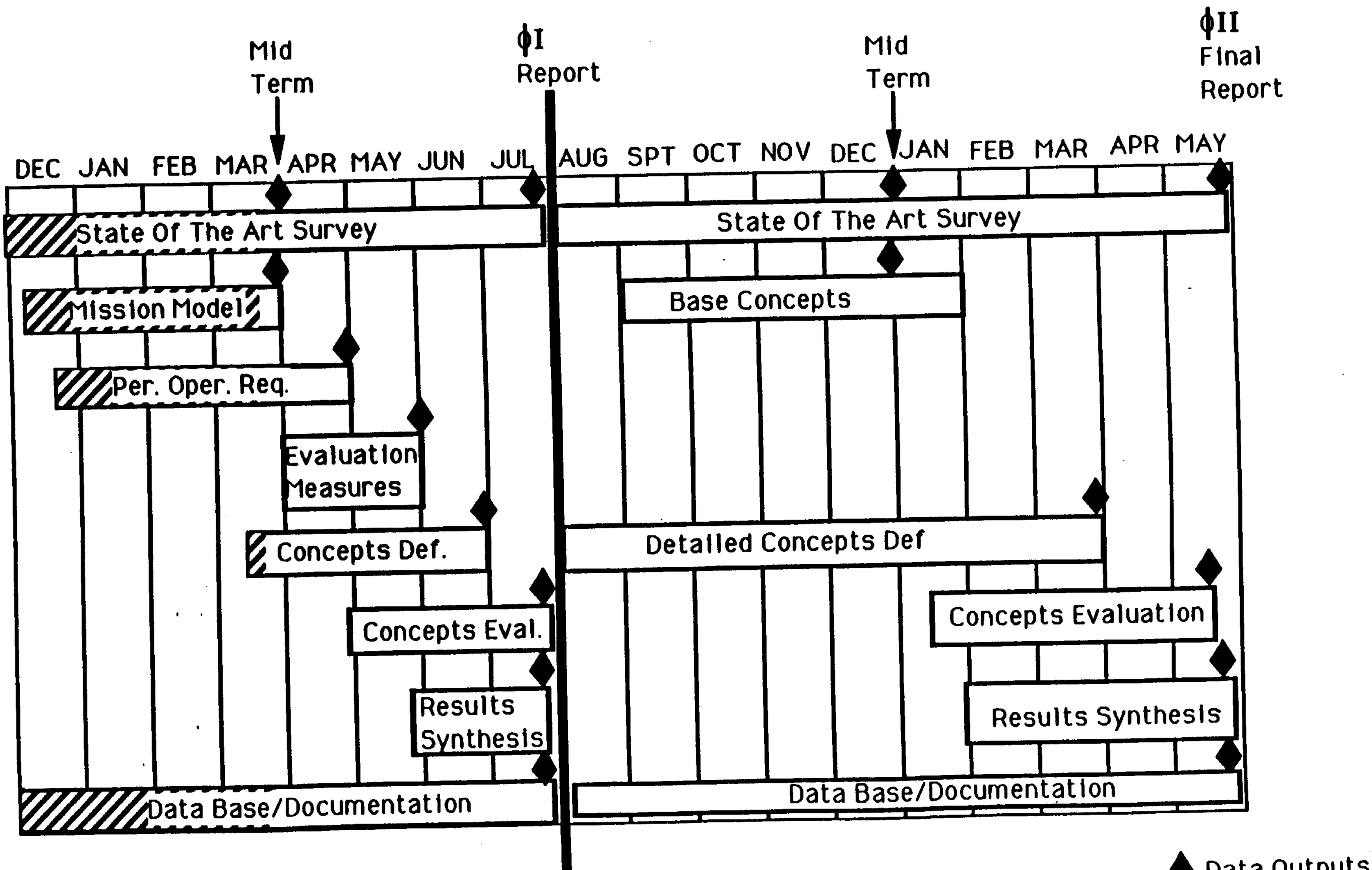

FIG 1

Piloted Rovers Task

This chart presents the accomplishments in the three (3) tasks undertaken in this period.

The mission model task definition and analysis continues to support the need for state of the art improvements over the technologies involved in the Apollo Lunar Vehicles.

A detailed vehicle mission model had been defined based on the NASA 90 Day Study.

Performance/operations requirements are being identified for MSFC review and comment following the presentation scheduled for March 30, 1990.

Piloted Rovers Task

Accomplishments

State of the Art Survey:

Improvements in Wheel Structural Design Required

Advanced Technology Wheel Drive Motors need to be incorporated

Rechargeable Batteries and Recharge Systems Required

Astronaut-to-Vehicle and Vehicle-to-Base Communication Improvements Required

Mission Model:

A Baseline Set of Vehicle Tasks for missions 0 through 8 have been defined

Performance/Operations Requirements Definition

Vehicle Requirements are being identified

Payload weight and size
Distance traversed and time to perform tasks
Duty Cycles (charge/recharge requirements)

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Programmatics Task 8.0

Agenda

WBS / WBS Dictionary Status
Top Level WBS Update
Lunar Mission Vehicle WBS Update
Cost Estimation Ground Rules
Cost Analyses
Element / Mission Schedules

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WBS / WBS Dictionary Status

WBS

- Update to Dec 89 Draft in Briefing Book
- Final Draft to NASA 6/5/90
- Final Delivery With Final Report

WBS Dictionary

- Preliminary 30% Complete
- Preliminary Delivery to NASA 7/2/90
- Final Delivery With Final Report

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Lunar Mission Vehicle WBS Update

Lunar Mission Vehicle

Management
- PP & C
- Finance
- Procurement
- Facilities
- Associate Contractor Mgt
- Inventory
- Laboratories

System Engineering & Integration
- System Engineering
- System Integration
- Verification
- Configuration Management
- Information Management
- International Integration
- Simulation
- Special Studies

LTV
- Design Integration
- Structures/Mechanisms
- Avionics
  - • GN&C
  - • C&DH
  - • IAV
  - • MMI
  - • Other
- Thermal Control
- Electrical Power
- ECLSS
- Manned Systems
- Software
- Support Equipment
- Test Article Integration, Assembly & Test
- Flight Article Integration, Assembly & Test
- Recovery System
- Manufacturing & Assembly

LEV
- Design Integration
- Structures/Mechanisms
- Avionics
  - • GN&C
  - • C&DH
  - • IAV
  - • MMI
  - • Other
- Thermal Control
- Electrical Power
- ECLSS
- Manned Systems
- Software
- Support Equipment
- Test Article Integration, Assembly & Test
- Flight Article Integration, Assembly & Test
- Recovery System
- Manufacturing & Assembly

Mission Assurance
- Safety
- Reliability
- Maintainability
- Quality Assurance
- S/W Product Assurance

Operations
- Pre-Launch
- Orbital
- Landing
- Post Launch
- Mission
- Logistics

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Cost Estimation Ground Rules

- Parametric for Hardware Elements
- Each Major Elements = Developmental Project
- ETFU for Developmental Program
- Cost/Mass Identification/Parametricization
- No Contingencies (NASA to provide)
- Payload Costs Factored
- No Learning Curve for < 4 units/yr
- 15% Initial Spares +10% of Active Mass/yr of Service for Reusable or Long Life
- Mission Ops Support Factored Only
- Ground Ops Factored From Hardware Cost
- Lumped Cost Spread (eg DD T&E))
- SE&I & Management Costs Factored
- In Space Support Factored
- Limited To Through STV Integration (eg SE&I)
    - No Mission
    - No Carry on
    - No Launch Vehicles (will use estimates for necessary cost trades)
- PRICE - S for Software
- PRICE - H for Hardware
- Ground Support - $/FT^2

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Cost Analyses

Preliminary Work on:

- **MMV DDT & E**
- **MMV Manufacturing**
- **MMV Support (Manufacturing & DDT & E)**
- **ECCV Unit Manufacturing by Subsystem**
- **TMIS/MTV Manufacturing by Subsystem**
- **MEV Manufacturing by Subsystem**

First Release Scheduled for May 30, 1990

Mars Vehicle Hardware DDT&E Cost Summary

Cost distribution summaries are presented on the next four pages. These are preliminary results; updates and refinements are expected to increase the totals but not greatly affect the general distributions.

Mars Vehicle Hardware DDT&E Cost Summary

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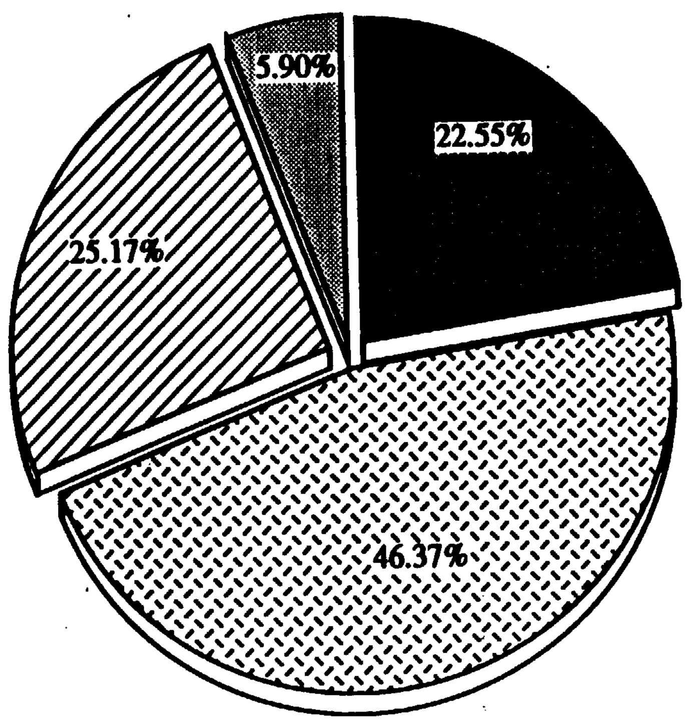
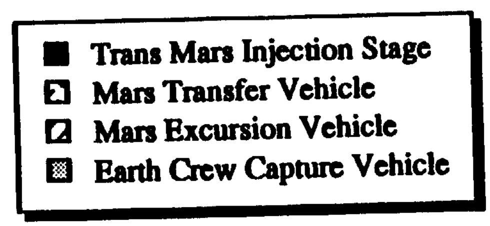


Total Cost = 2634.1 M



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ADVANCED CIVIL SPACE SYSTEMS

Mars Vehicle Hardware Manufacturing Cost Summary

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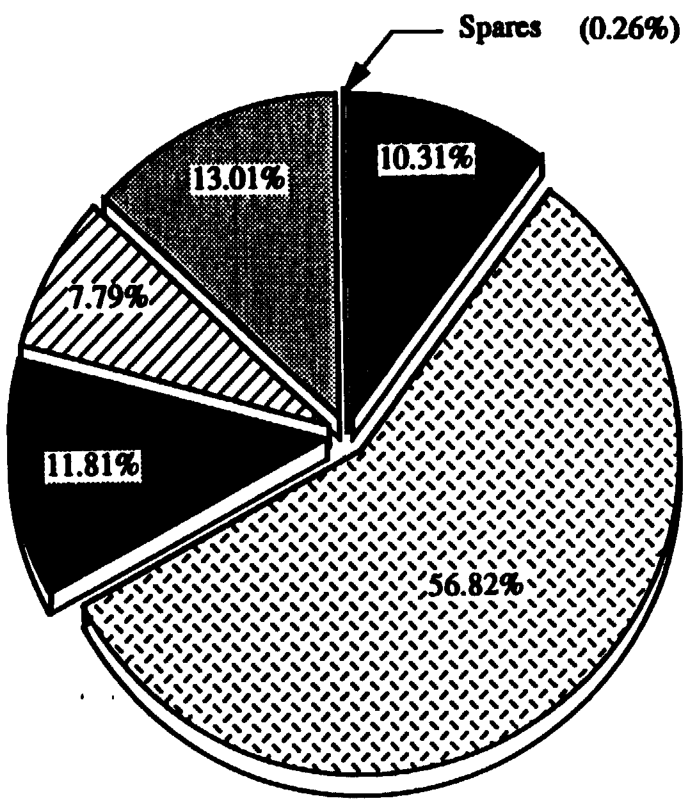
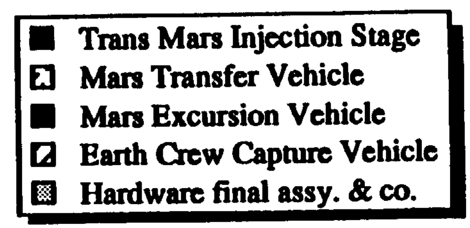


Total Cost = 3154.87 M



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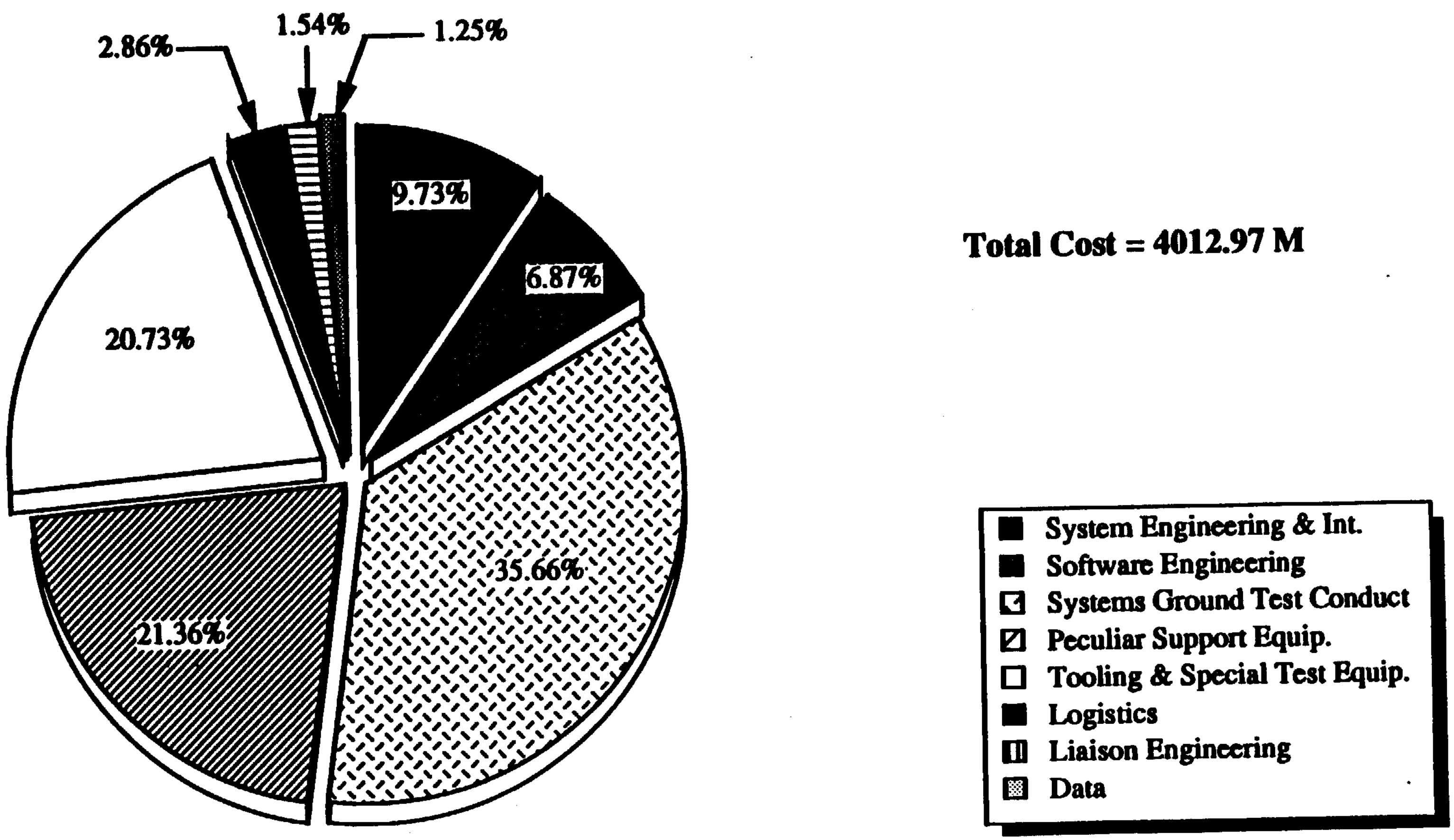
Mars Vehicle Total Support Cost Summary



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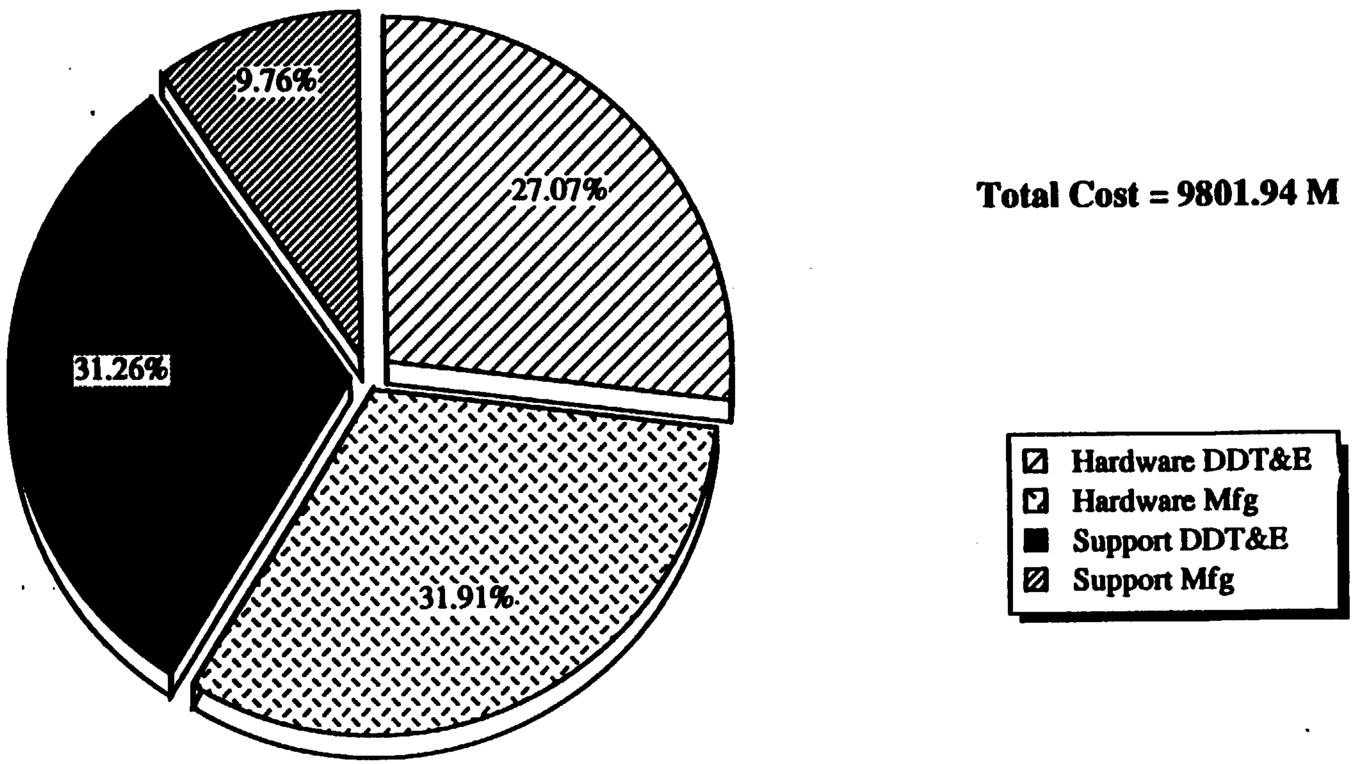
Mars Vehicle Total Cost (Hardware & Support)



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Element/Mission Schedules

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- **Preliminary Work On:**
  - **Missions**
  - **Propulsion**
  - **Aerobrakes**
  - **Habitat Module**
  - **Support Systems**
  - **Rovers**

- **First Release Scheduled for April 30, 1990**

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Evolutionary and Innovative Architectures

Gordon Woodcock

ADVANCED CIVIL SPACE SYSTEMS

Agenda

Candidate Alternative Architectures
Mars Transportation Architecture Options
Evolutionary Program Commonality Matrix
Preliminary Architecture Schedule and Manifest
Overarching Goals for Evolutionary Architecture
Objectives of Evolutionary Architecture Analysis
Lunar / Mars Sep Transportation Infrastructure
Innovative / Evolutionary Architecture Themes

Objectives of Evolutionary Architecture Analysis

The objectives of evolutionary architecture analysis are listed on the facing page. The overall purpose is to cast the main transfer vehicle technologies in representative architectures in order to obtain valid comparisons and evaluations of advantages, disadvantages and life cycle cost.

Objectives of Evolutionary Architecture Analysis

- Develop alternative architectures that incorporate evolution, innovation and advanced technology.
- Develop and exercise a methodology for synthesizing and optimizing program architectures.
  - Synthesis method
  - Evaluation criteria
  - Analysis methods
  - Model sets of architecture definition data
- Accomplish definitive broad trades that can only be resolved by embedding in alternative system/ program architectures.
- Assist in developing an overall technology advancement strategy.

Overarching Goals for Evolutionary Architectures

Goals are stated here. These, in effect, represent a particular program strategy that aims for early achievement of major milestones and evolves to advanced technologies capable of supporting human presences on the Moon and Mars on a larger scale than represented in the NASA "90-day Study".

Overarching Goals for Evolutionary Architectures

- Minimize the total cost of reaching initial program goals, i.e. manned return to the Moon, thereby enabling earliest possible accomplishment.

- Maximize long-term accomplishment benefit/cost ratio, such as person-years on planetary surfaces per unit of program life-cycle cost.

Innovative/Evolutionary Architecture Themes

The program strategy and goals from the previous page lead to the themes stated here. These themes were used as guide-posts in formulating architectures.

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Innovative/Evolutionary Architecture Themes

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- Minimize number of development projects to achieve first program goals; aim for early achievement.
- Each new development project should add significant capability or enable new program goal.
- Aim for architectures capable of evolutionary growth to support large bases or proto-settlements.
- Cost & economics-driven top-down synthesis of architectures across total program.
- All requirements below program goals justified by design, analysis, and trades. Design by analysis, design synthesis, and trade studies, not by writing requirements.
- Safety and reliability items viewed as standards. (Not derived requirements.)
- "Use everything but the squeal" approach to commonality and multi-purpose equipment.
- New technology justified by literal necessity or high rank according to ROI analysis.
- No new technology is off limits if it can be quantified on engineering/scientific principles. (For example, mass drivers and ram accelerators are much more quantifiable than gas-core rocket.)

C-7

Candidate Alternative Architectures with Focussed Themes and Premises

The next three pages present seven alternative architectures (alternative to the cryogenic/aero-braking reference). Each one has a set of premises and themes. The reasons for selecting these architectures are to test the premises and compare results.

The "Bolo" idea was placed in the lunar architecture because it is innovative, and provides a major reduction in lunar mission profile delta V budget. Results of mode performance analysis, however, show this mode to have relatively little advantage over use of lunar oxygen in the LOR mode. Therefore, we will test the value of the bolo concept by taking it out of some of these architectures.

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Candidate Alternative Architectures with Focussed Themes and Premises

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Architecture	Themes, Premises
1. Lunar - Cryo. aerobrake, evolving to nuclear surface power, lunar oxygen, CELSS, and Bolo Mars - Early lunar-derived mission evolving to NEP and reusable MEV with Mars surface oxygen.	(a) Early manned lunar mission at minimum cost using only LTV; (b) early evolution to LTV/LEV, lunar ISRU, 2 - year stay time, and CELSS minimizes resupply, opening "Mars wedge"; (c) Early Mars mission is important, can be done using modified lunar hardware, "easy" 2010 opportunity and L2 basing; (d) Permanent Mars transportation can be efficient and fast using NEP and reusable MEV with Mars surface oxygen; (e) NEP technology base is synergistic with planet surface power base; (f) Lunar orbit bolo reduces ΔV by ~ 3500 m/sec, good investment later with growing lunar base.
2. Lunar - Cryo- Aerobrake, evolving to NTR LTV; nuclear vs. beam surface power options; CELSS; LLOX for LTV Mars - NTR with options for early lunar-derived mission and reusable MEV as in #1.	(a) NTR performance payoff for lunar use; (b) lunar mission gets early NTR operational experience; (c) beam surface power option avoids supporting two space nuclear technology infrastructures; (d) tests NTR vs. NEP for Mars.

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Candidate Alternative Architectures with Focussed Themes and Premises p. 2

Architecture	Themes, Premises
3. Lunar - Cryo. aerobrake, evolving to beamed surface power, lunar oxygen, CELSS, and Bolo Mars - Early lunar-derived mission evolving to SEP and reusable MEV with Mars surface oxygen.	This alternative is like #1, but with beamed power substituted for nuclear planet surface power, and SEP substituted for NEP. This case tests the costs versus benefits of nuclear space electric power. Cost of large-scale solar arrays is treated parametrically so that the range of solar/nuclear crossover is defined.
4. Lunar - Cryo. aerobrake, evolving to nuclear surface power, lunar oxygen, CELSS, and Bolo Mars - Cryo. aerobrake with options for early lunar-derived mission, evolving to L2 Mars node supplied with lunar oxygen by lunar surface mass driver.	This alternative is like #1, but evaluates a Mars option emphasizing lunar resources in place of advanced propulsion. The L2 node makes supply of lunar oxygen for the Mars mission efficient, and minimizes resupply from Earth. Depending on the lunar biomass balance, and the availability of volatiles resources on the Moon, MTV waste disposal on the Moon and food and atmosphere resupply from the Moon may also make sense.

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ADVANCED CIVIL SPACE SYSTEMS

Candidate Alternative Architectures with Focussed Themes and Premises p. 3

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Architecture	Themes, Premises
5. Lunar - Cryo. aerobrake, evolving to nuclear surface power, lunar oxygen, CELSS, and Bolo Mars - This alternative retains early Mars option as #1, but evolves to a pair of cycler orbit stations with cryo. aerobrake taxis for Earth/Mars encounters.	This alternative tests the technical feasibility and costs versus benefits of the cycler orbit station concept, using up- and down- escalator orbits, with solar electric propulsion as necessary to propagate the orbits.
6. Lunar - Cryo. aerobrake, evolving to nuclear surface power, lunar oxygen, CELSS, and Bolo Mars - Option for early lunar- derived mission; evolving to integrated fully reusable MTV/MEV, refueled with both hydrogen and oxygen on Mars surface for return to Earth orbit. This integrated vehicle flies only conjunction profiles ($\Delta V \leq 7$ km/sec).	This alternative simplifies the transportation architecture by relying on mainly robotic buildup of a Mars surface infrastructure including propellant production. Hydrogen production will require a source of tonnes of water. Expedition-class piloted Mars missions during the infrastructure buildup would use lunar-derived systems.
7. Lunar/Mars - Identical to option 1, but uses NEP for lunar cargo delivery.	This alternative tests the cost-effectiveness of NEP for lunar cargo where the NEP is already available.

Mars Transportation Propulsion Options for Alternative Architectures

A key factor in defining and selecting propulsion options is the pertinent operations modes. Operations mode principal factors are the degree of reusability, the location and character of the transportation node, and the means of fueling and refueling. An option matrix was created to categorize transportation options according to these factors. The degree of reusability is indicated on the left, as part of the type description, and the other factors are used as table headers. Numbers in the table are architecture option numbers.

Options 1 and 2 are based on the reference system. Option 1 is the reference, and option 2 is a variant that is assembled off Space Station Freedom. The rationale for off-station assembly is to minimize interference among lunar and Mars operations and other space station operations.

Options 3, 4, and 5 represent two different ways of making the reference system fully reusable. Here, the major reuse item is the TMIS part of the MTV element; it is expendable in the reference case. Options 3 and 4 represent two ways of achieving reuse; a staged TMIS and use of an L2 node. Option 5 is a variation on option 3 where an orbiting propellant depot is used; in options 3 and 4 the tankers deliver direct to the Mars vehicle. Staging the TMIS at slightly less than Earth escape energy or use of an L2 node divides or reduces the TMI delta V. The staged element of the TMIS enters an elliptic orbit from which it can return to LEO by aerobraking. The balance of the TMI delta V, about 1200 m/sec, is allocated to the Earth-Mars-Earth MTV stage. This may be accommodated by enlarging the MTV tanks or by drop tanks. Use of L2 has a similar effect; the TMI delta V from L2 is about 1500 m/sec via powered Moon/Earth gravity assist. There is no large TMIS. Upon return to Earth, the MTV enters a translunar trajectory rather than aerobraking to low Earth orbit. This serves, among other things, to significantly reduce Earth capture aeroheating. Return to L2 from the translunar trajectory requires slightly over 300 m/sec.

A sub-option involves whether the MEV is reused. A reusable MEV requires surface refueling with Mars-produced oxygen for ascent to be practical. Reuse of the MEV implies

return to the same Mars orbit on every mission, a flight mechanics constraint not otherwise imposed. Reuse of the MEV imposes mission delta V penalties yet to be determined.

A further variation on option 4 uses lunar oxygen to refuel the MTV at L2. Liquid hydrogen comes from Earth. Delivery of lunar oxygen to L2 can be accomplished by conventional cryogenic rocket or by mass driver, the latter using oxygen tanker cannisters. This was viewed as different enough to represent a distinct mission architecture, alternative #4.

Options 6 and 7 are the conventional nuclear thermal rocket scenario, embedded in alternative architecture #2. Option 7 uses a high-Isp, low pressure, low thrust NTR. These options are all propulsive; one of their benefits is elimination of development of high-energy aerobraking (an aerobrake is still needed for Mars landing). These options include sub-options as described on a subsequent page, in which the mode of return to Earth is varied. The NTR uses hydrogen propellant. Large, insulated liquid hydrogen tanks weigh about 18% of their hydrogen contents. Unless staged, they introduce a large inert mass penalty that makes the NTR at 900 Isp not an attractive option.

Options 8 and 9 are fully reusable; all of the propellant tanks are returned to Earth orbit for reuse. At 1250 Isp or above, complete reuse is a viable option. We selected 2500 Isp as representative of the nuclear gas-core (NGC) system; if the gas-core engine works, this level of performance is generally accepted as achievable.

Options 10, 11, and 12 are variations on solar electric propulsion. Options 10 and 12 are assembled at Space Station Freedom and delivered to L2 using a sacrificial solar array that is discarded after degradation by spiraling through the van Allen belts. Option 11 is delivered to L2 in subassembly packages by a lunar transfer vehicle (LTV) and deployed or assembled there. Option 12 uses tank exchange for refueling (wiht liquid argon propellant); options 10 and 11 use tankers delivered to L2 by LTV. Once at L2, the SEP never returns to LEO. Subsequent missions depart from L2. Crew, resupply propellant, and cargo are delivered to L2 by LTV.

Option 13 involves operation of a nuclear electric vehicle from the Space Station Freedom orbit. This option requires safety provisions to protect the space station crew and to provide high assurance that the power reactors, once used and therefore highly radioactive, cannot reenter Earth's atmosphere. The NEP spirals out of Earth orbit unmanned. Exposure to the van Allen belts is not deleterious to a nuclear powerplant; any sensitive items are protected by shielding. By the time the NEP is near lunar distance, it is about 2 days from Earth escape. At that time, an LTV delivers the crew to rendezvous and board the NEP (the delta V for this is about like that for a trip to low lunar orbit). In this way, the crew are not exposed to the van Allen belt spiral or to the long spiral time. Upon return from Mars, the reverse is done: an LTV meets the returning NEP at about lunar distance and brings the crew back to LEO. The NEP spirals down to LEO unmanned. It is parked at a distance of hundreds of kilometers from Space Station Freedom for about a month, after which its residual radioactivity has abated enough to bring it closer for turnaround operations.

Options 14 and 15 represent one way to deal with the nuclear safety-of-operations issue; the "hot" NEP is operated from a high-altitude node. In option 14, initial assembly and checkout are at Space Station Freedom (cold reactor, not a radiation hazard). The completed and checked out system is boosted to a nuclear-safe altitude of about 800 km by chemical means, after which it spirals to the node altitude under its own power to begin regular operations. The NEP operates from and returns to a high-orbit node at, for example, GEO or L2, and is serviced there by an LTV or by a beamed-power STV. In option 15, the NEP is assembled and checked out at the high orbit node after delivery there in sub-assemblies by an LTV/STV. The NEP, like the SEP, uses liquid argon propellant.

A further consideration for options 13 - 15 is disposal of the spent NEP reactors. A typical reactor lifetime expectation will serve 5 to 10 round trips to Mars, after which the reactor must be disposed of. Options include NEP self-power (the entire vehicle) to a safe parking orbit such as 0.85 a.u. circular solar orbit, delivery of the spent reactor only to safe disposal by an LTV, or use of multiple gravity assists on return from the last mission to dispose of the system by solar system escape.

ADVANCED CIVIL SPACE SYSTEMS

Mars Transportation Architecture Options

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Propulsion/ Vehicle Type	NODE(S) Assy at SSF	Assy off SSF	HEO/L2 Node	SSF Support	FUELING/REFUELING Launch Loaded	Tankers Direct	Tankers to Depot to Vehicle
Cryo/Aerobrake Partially Reusable (Reference)	(1)	(2)		(2)	(1 & 2)		
Cryo/Aerobrake Fully Reusable	(3 &5)		(4)	(4)		(3 & 4)	(5)
NTR 900 Isp Staged Tanks & Engines	(6)				(6)		
NTR 1250 Isp Staged Tanks & Engines	(7)				(7)		
NTR 1250 Isp Fully Reusable		(8)	(8)	(8)		(8)	
GCR Isp 2500 Fully Reusable		(9)	(9)	(9)		(9)	
SEP Operated from L2	(10 &12)	(11)	(11)	(11)	(12)	(10 & 11)	
NEP Operated from SSF Orbit	(13)				(13)		
NEP Operated from High Orbit/L2	(14)	(15)	(15)	(15)	(14)	(15)	

Fully Reusable Cryogenic Aerobraked System, Split TMI Burn (Modes 3 and 5)

The most practical way to make the TMI stage reusable is to split it: most of the delta V is delivered by a TMI booster which separates slightly below escape energy, typically at about lunar transfer conditions. After one revolution of about 5 days' duration, the TMI booster recaptures into low Earth orbit. The balance of the TMI delta V is delivered at the first perigee, using propellant capacity on the MTV transit vehicle. This propellant capacity could either be incorporated in the MTV tanks, or in drop tanks sized for the TMI second burn. (In the latter case, the TMI system is not quite fully reusable, but the expended tanks are small and contain no engines or avionics.)

This split-burn strategy avoids necessity to "turn around" a large TMI stage at Mars transfer C3 and return it to Earth orbit in order to reuse it. The turnaround delta V is typically twice or more the delta V for the MTV transit burn from a 5-day ellipse to TMI, because by the time the turnaround maneuver can be executed, the TMI stage is several thousand km. altitude above low Earth orbit.

Fully Reusable Cryogenic Aerobraked System, Split TMI Burn (Modes 3 and 5)

ADVANCED CIVIL SPACE SYSTEMS

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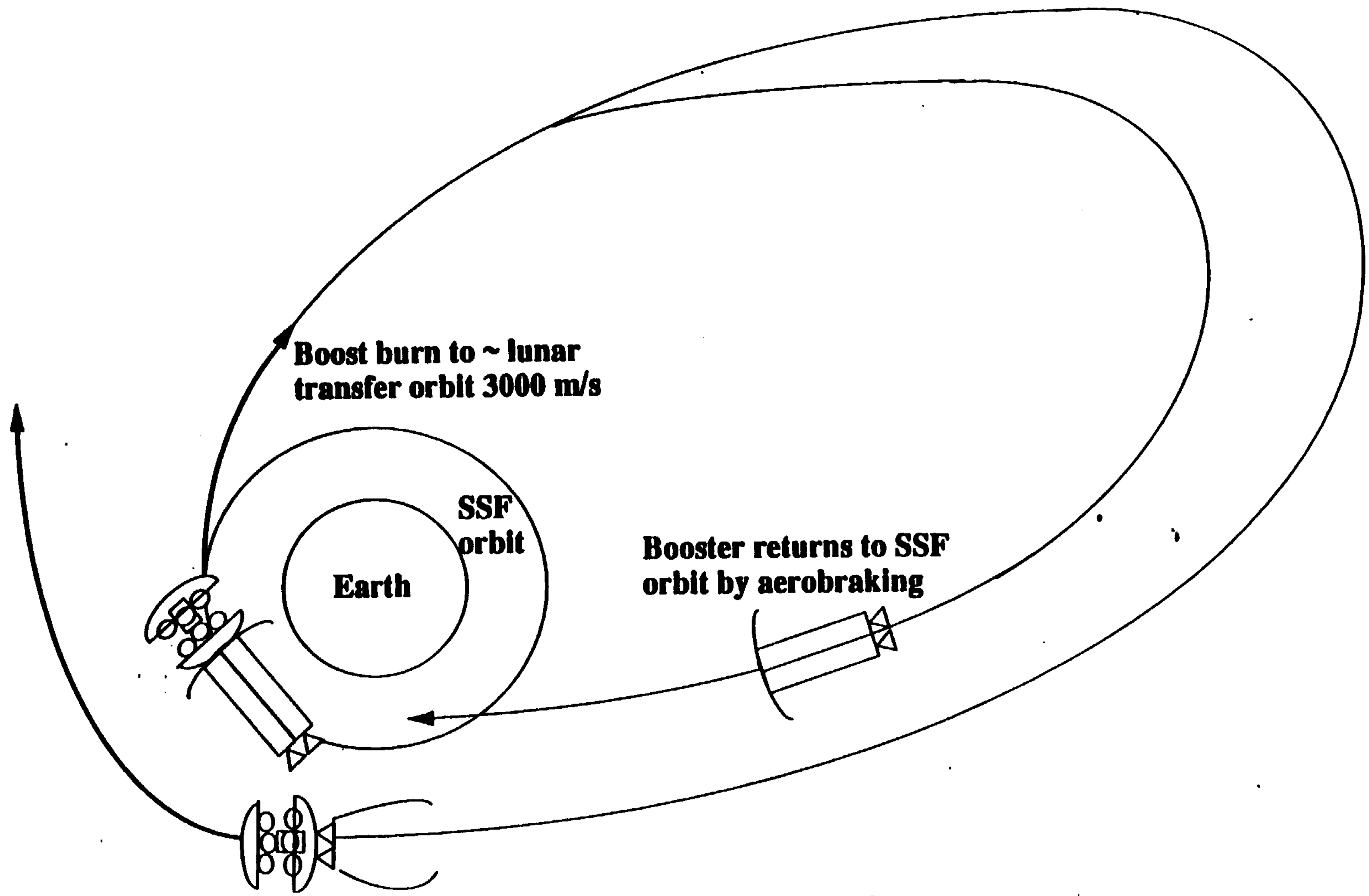

TMI burn ~ 1200 m/s, uses MTV propulsion with extra tanks.
MEV delivered to Mars and reused there; MTV returns to SSF orbit

NTR 900 Isp Staged Tanks and Engines, Mode 6

This chart depicts the operating modes for a Nerva-type nuclear thermal rocket (NTR). At 900 Isp, it is necessary to stage depleted propellant tanks to obtain reasonable performance. At 1250 Isp, a fully reusable mode is possible within an acceptable range for initial mass in Earth orbit (IMLEO).

The NTR vehicle performs a single "burn" to trans-Mars injection. At this point, the propellant tanks for TMI, about half the initial vehicle mass when loaded, are jettisoned. Engines may be jettisoned with these tanks since the thrust level needed for efficient TMI is much greater than needed on the rest of the mission. The core stage performs Mars orbit insertion (MOI), after which more tanks (but not engines) are jettisoned. Mars orbit operations are the same as for the cryogenic/aerobrake reference. Upon completion of Mars operations, the core stage performs trans-Earth injection (TEI).

There are three options for Earth return. Mode 6A is the most common assumption; the only part of the mission system returning to Earth is an Earth crew capture vehicle (ECCV), which may return either to Earth orbit by aerocapture, or direct to the surface a la Apollo. Mode 6B saves the NTR core stage and crew transit habitat for reuse by propulsively capturing it into Earth orbit. The initial capture is into an orbit higher than the SSF orbit (typically 800-1000 km); the crew are returned to SSF by an LTV sortie. After 30 days of reactor cooldown, the NTR is returned to the SSF orbit for reuse.

Mode 6C captures the core stage and habitat into a high Earth orbit such as GEO or L2; the crew separates from the core stage a few days before Earth arrival and returns by ECCV.

NTR 900 Isp Staged Tanks and Engines, Mode 6

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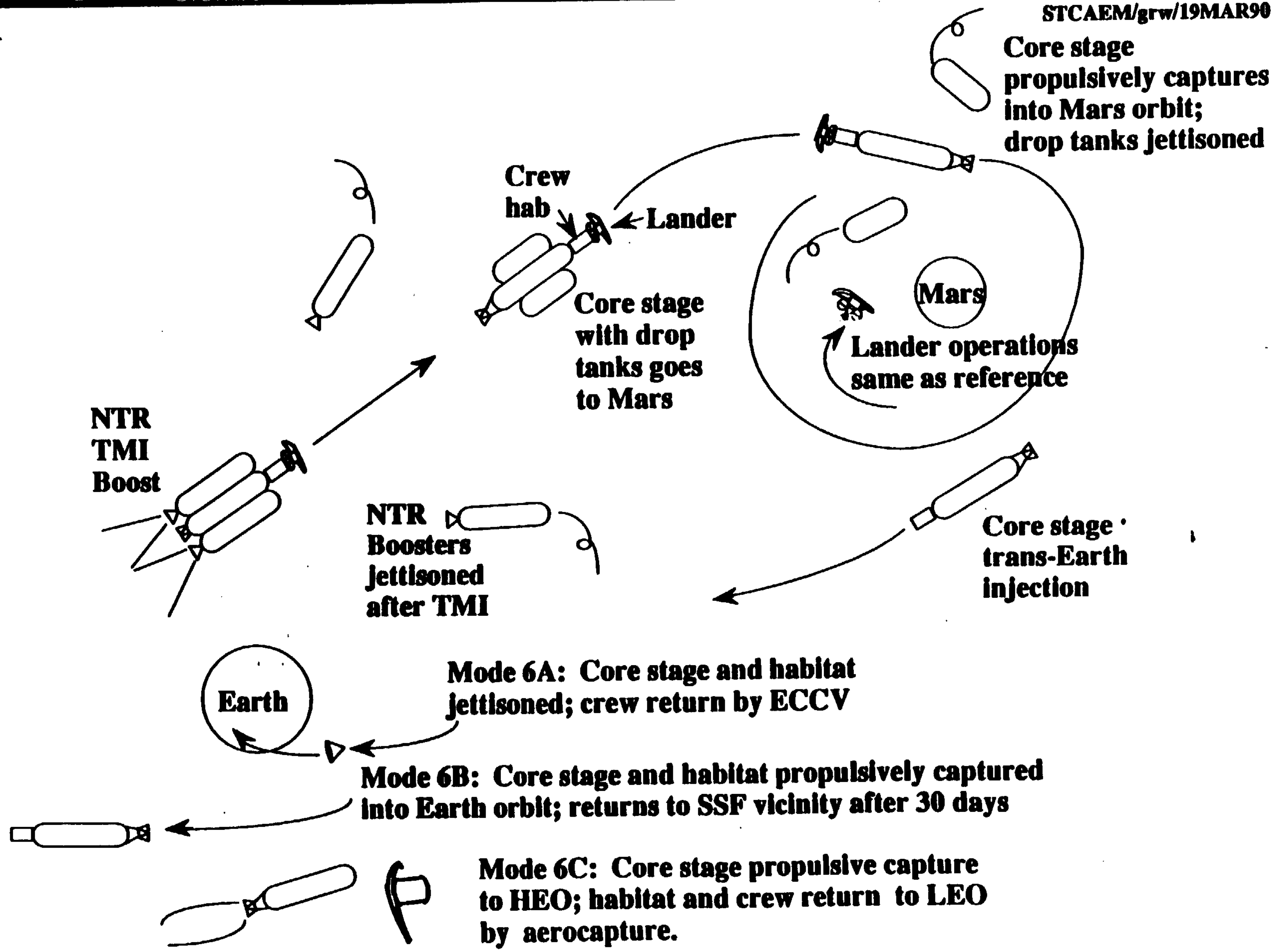

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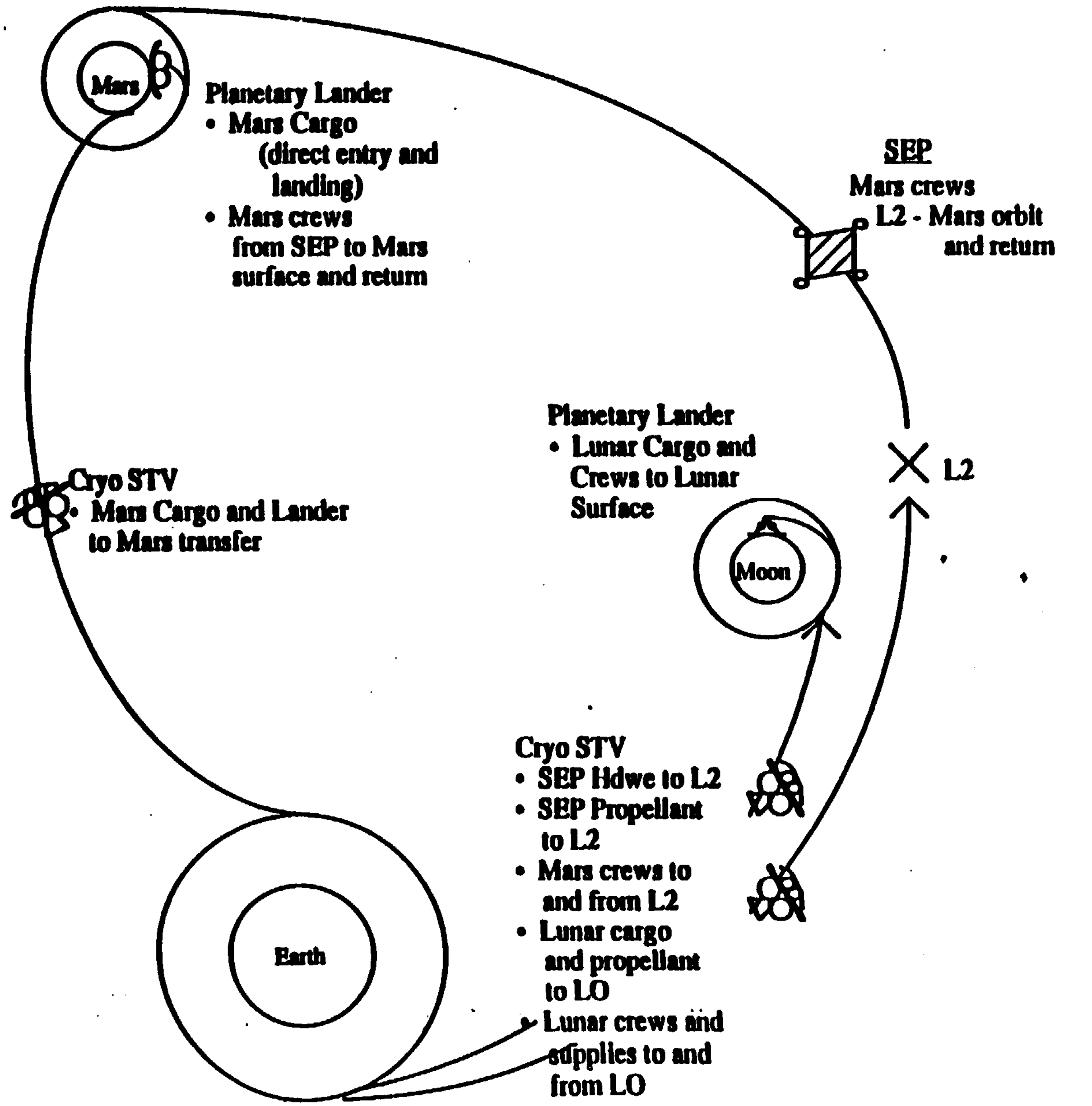
Lunar/Mars SEP Transportation Infrastructure



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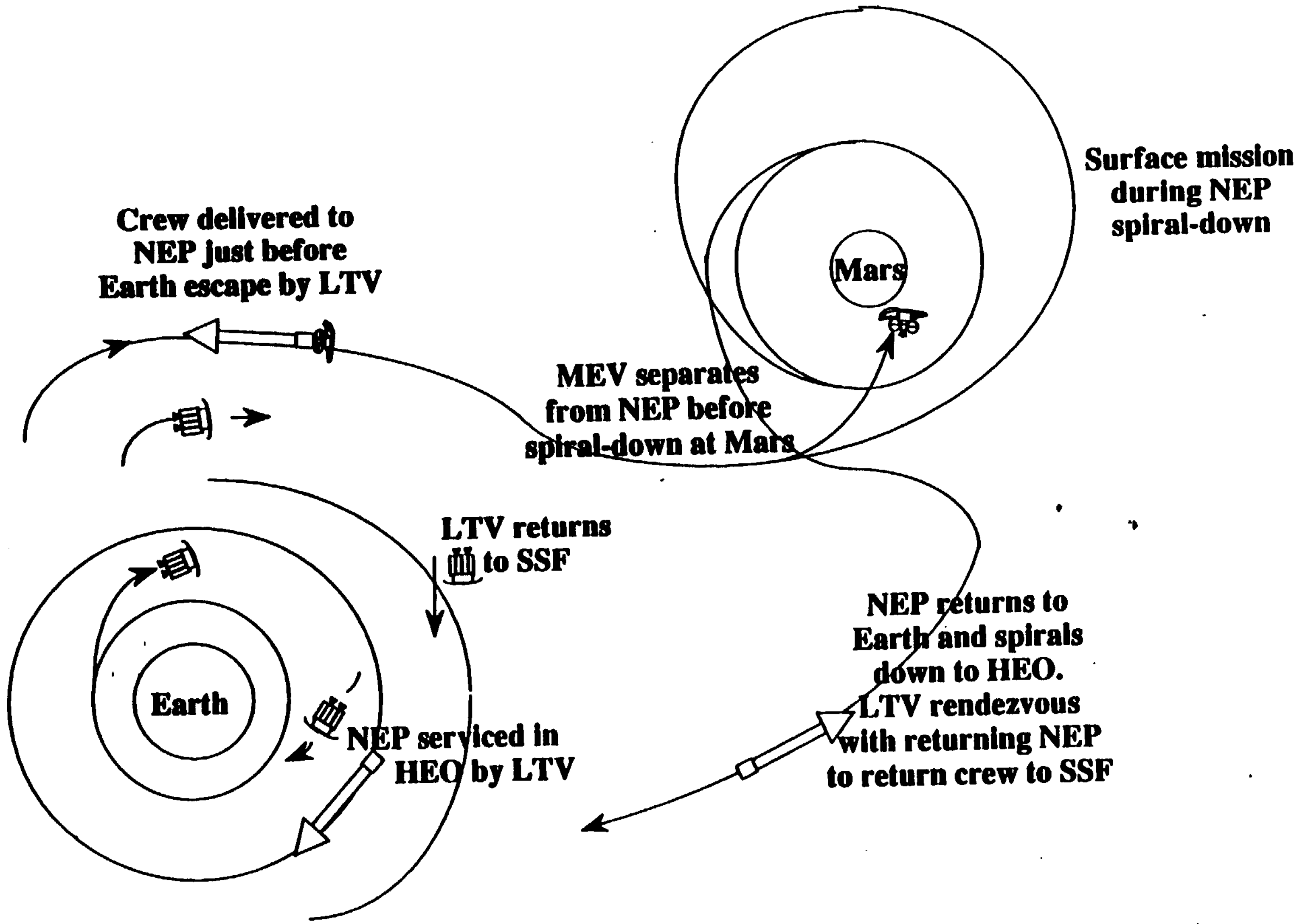
NEP Operated from High Orbit, Modes 14 and 15

Operating an NEP or SEP from a high orbit reduces the lengthy spiral time needed to escape Earth from a low Earth orbit. If the high orbit is at GEO altitude, a few weeks spiral time may be needed; if at L2, escape occurs within one or two days. If a significant spiral time is needed, the crew can board the NEP just before escape. An LTV is used as a "taxi" to get to the NEP. The delta V for LEO to high orbit varies little between GEO and the Moon, and is actually less for L2 than for most other destinations.

The MEV separates from the NEP early in the spiral at Mars, or even before the spiral is initiated. During the surface mission the NEP spirals down into Mars' gravity well. Since the MEV ascent delta V also varies little with rendezvous altitude, there is little benefit to driving the NEP to low altitudes. Some benefit in reduced MEV delta V may accrue from making the NEP parking orbit elliptic.

After the surface mission is complete and the MEV returned to Mars orbit, the NEP spirals out to Earth transfer. Upon approaching Earth, an LTV "taxi" meets the NEP and returns the crew to LEO, while the NEP returns to its high orbit parking location.

NEP Operated from High Orbit, Modes 14 and 15



Preliminary Architecture Schedule and Manifest

The figure shows a preliminary schedule for alternative architecture #1. Development projects are phased to introduce a new capability every year or two. The initial return to the Moon is accomplished with a tandem/direct LTV, requiring only two system integration projects, Shuttle-C and the LTV, and 10 sub-element development projects. These systems are retained through the life of the program.

The program achieves an early first visit to Mars by exploiting the low-energty 2010 opportunity, using a modified lunar LTV and LEV for Mars, and staging from L2 to reduce the propellant load for trans-Mars injection to a value compatible with the LTV. This presumes a favorable solution to the long-term zero g problem, i.e. soon after the year 2000, in time to carry out the mission.

High productivity is achieved through (1) evolution to the most efficient lunar transportation modes, (2) evolution to a significant degree of lunar surface self-sufficiency through ISRU, and (3) evolution to the NEP, Mars surface oxygen and a reusable MEV for Mars missions, with about 1/3 the LEO resupply requirement of the reference cryogenic/aerobraking system. This leads to a projection of up to 36 people on the Moon and 18 on Mars by 2025.

Preliminary Architecture Schedule and Manifest

ADVANCED CIVIL SPACE SYSTEMS

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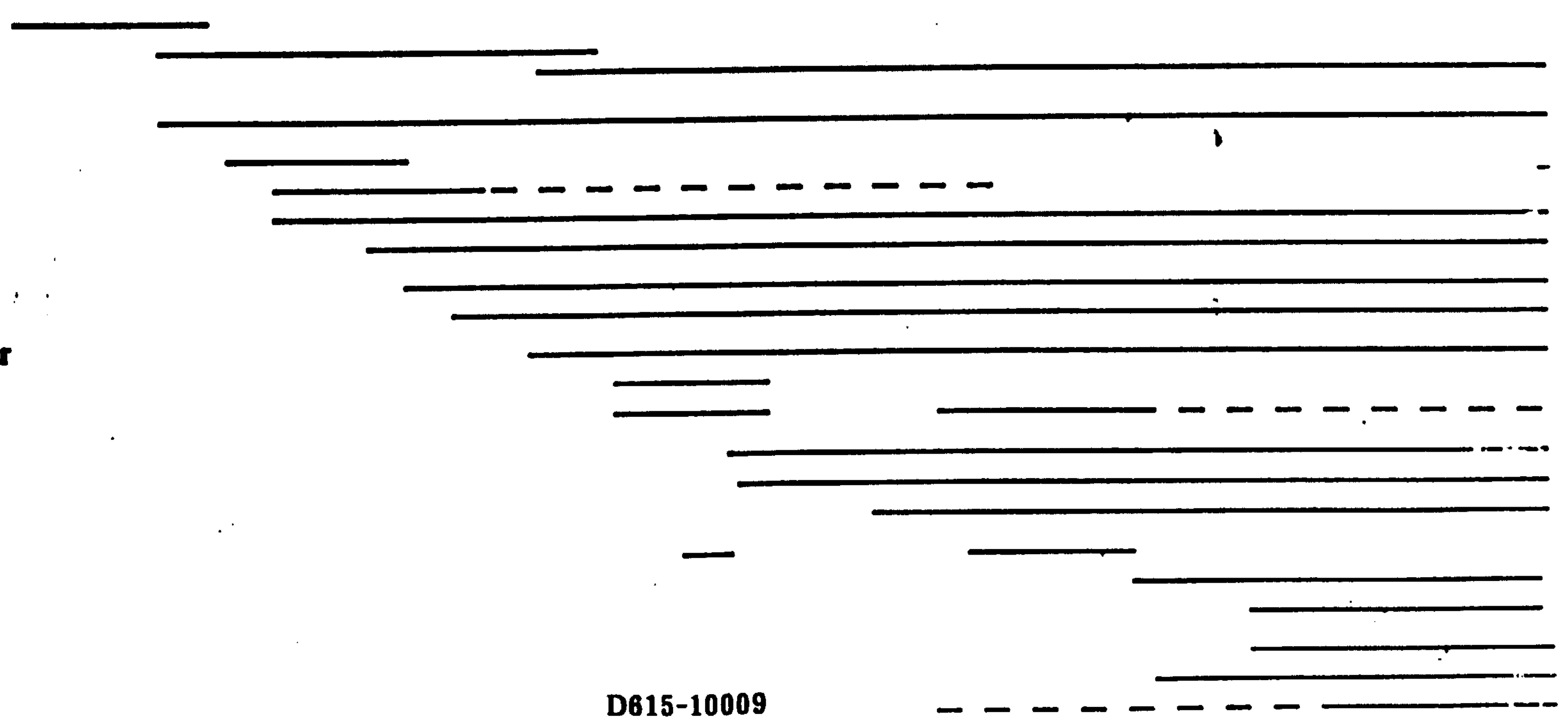
	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27
Shuttle - C/I	2	2	2	2																												
II					4	4	6	6	7	7	7	7	7	2																		
III															6	6	5	5	5	4	6	6	6	6	6	6	5	6	6	6	6	6
Lunar Crew Trips					1		1	1	1	1	1	1	1	1	1	2	1	1	1	1	2	2	2	2	2	2	2	3	3	3	3	3
Lunar Cargo Trips						1		1	1	1	1	1	1			1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1
Mars Crew/Cargo															1			C		1			1		1		1		1		1	
Lunar Pop.					4/12d		4/54d		6/6m	6	6	12	6	6	6	12	12	12	12	12	24	24	24	24	24	24	24	30	36	36	36	36
Mars Pop.															5/20d					6/60d			6/60d		6	6	12	12	12	12	18	18

System Integration Projects



Shuttle-C Block I
Shuttle-C Block II
Shuttle-C Block III
SSF Node
Tandem LTV
Lunar Campsite
Lunár Open Rover
Lunar Constr. Equip.
LTV/LEV
Lunar Base
Lunar Pressurized Rover
MTV (L2 Based)
Mini-MEV
Industrial Lunar Power
Lunar ISRU
Mars NEP
Mars Campsite
Mars Base
Industrial Mars Power
Mars ISRU
Reusable MEV
Lunar Orbit Bolo

Evolutionary Program Commonality Matrix

This figure illlustrates a matrix method for formulating an architecture with high commonality. The matrix represents alternative architecture #1 (NEP). The matrix presumes a project organization where major system sub-elements, such as crew modules, engines, and power generators, are procured as hardware/software development projects, and these are then used by systems integration projects that field complete integrated systems such as LTVs and lunar surface powerplants.

The level of commonality achieved could be exploited to dual-source certain key sub-elements, maintaining a competitive environment through the life of the program.

Evolutionary Program Commonality Matrix

ADVANCED CIVIL SPACE SYSTEMS

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Program Phases

System Integration Projects	SSF	Lunar Emplacement	Lunar Consolidation	Lunar Utilization	Mars Emplacement	Mars Consolidation	Mars Utilization
Shuttle-C Block I	x						
Shuttle-C Block II		x	—	—			
Shuttle-C Block III				x	—	—	—
SSF Node		x	—	—	—	—	—
LTV		x	—	—	—	—	—
Lunar Campsite		x	—	—	—	—	—
Lunar Open Rover		x	—	—	—	—	—
Lunar Constr. Equip.			x	—	—	—	—
LEV			x	—	—	—	—
Lunar Base			x	—	—	—	—
Lunar Pressurized Rover			x	—	—	—	—
MTV (L2 Based)					x		
Mini-MEV					x	—	—
Industrial Lunar Power				x	—	—	—
Lunar ISRU				x	—	—	—
Mars NEP						x	—
Mars Campsite						x	—
Mars Base						x	—
Industrial Mars Power							x
Mars ISRU							x
Reusable MEV							x
Lunar Orbit Bolo				x			

Subsystem Projects (Building Blocks)

System Integration Projects	Shuttle-C 4.5 m Cargo Carrier	Shuttle-C 7.6 m Cargo Carrier	Shuttle-C 10-m Cargo Carrier	STME 585k	LRB??	Recov. Prop/Avionics Module	Trans Vehicle Avionics Suite	LEO Tanker	Ass'y Node Debris Shield	Adv Space Eng 30k	LTV Stage	Lunar Aerobrake	LTV/Surface Hab	Solar/RFC Power Units	Space & Lunar Surface TCS	Small Rover Chassis	Small Wheel & Drive System	Large Rover Chassis	Straddler Chassis	Large Wheel & Drive System	LEV Stage	LEV Crew Cab	Mars Aerobrake	MTV/Surface Lg Dia Hab	Multi-Megawatt Reactor	Potassium Turbogenerator	Powerplant Heat Rej. System	Multi-Megawatt PMAD Equip	Lunar ISRU Equipment	Mars ISRU Equipment	Lunar Orbit Bolo Str & Mech
Shuttle-C Block I	x						x	x																							
Shuttle-C Block II		x				x	x	x																							
Shuttle-C Block III	x	x	x	x	?	x	x	x																							
SSF Node									x																						
LTV							x			x	x	x	x	x	x																
Lunar Campsite													x	x	x																
Lunar Open Rover																x	x														
Lunar Constr. Equip.															x	x	x	x	x	x											
LEV							x			x											x	x									
Lunar Base													x	x	x									x							
Lunar Pressurized Rover														x	x			x		x		x									
MTV (L2 Based)							x			x	x			x	x								x	x							
Mini-MEV							x			x											x	x	x								
Industrial Lunar Power																									x	x	x	x			
Lunar ISRU																													x		
Mars NEP							x																	x	x	x	x	x			
Mars Campsite													x	x	x																
Mars Base													x	x	x	x	x	x	x	x				x							
Industrial Mars Power																									x	x	x	x			
Mars ISRU																														x	
Reusable MEV							x			x					x							?									
Lunar Orbit Bolo							x							x	x																x

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Next Three Months Highlights

The facing page provides a "preview" of the next quarterly report.

Next Three Months Highlights

- Finish aerobraking analyses.
- Finish mission profile analyses.
- Finish advanced propulsion configuration concepts.
- Major emphasis on evolutionary architectures, operations, and programmatics.

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16. Abstract

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